NASA Contractor Report 195427/VOL2

High Pressure, Earth-Storable Rocket Technology
Volume 2—Appendices A and B

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Prepared under Contract NAS3–27003

National Aeronautics and
Space Administration

Lewis Research Center

October 1997

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APPENDIX A

TASK 1 – INFORMAL WRITTEN REPORT 9-93

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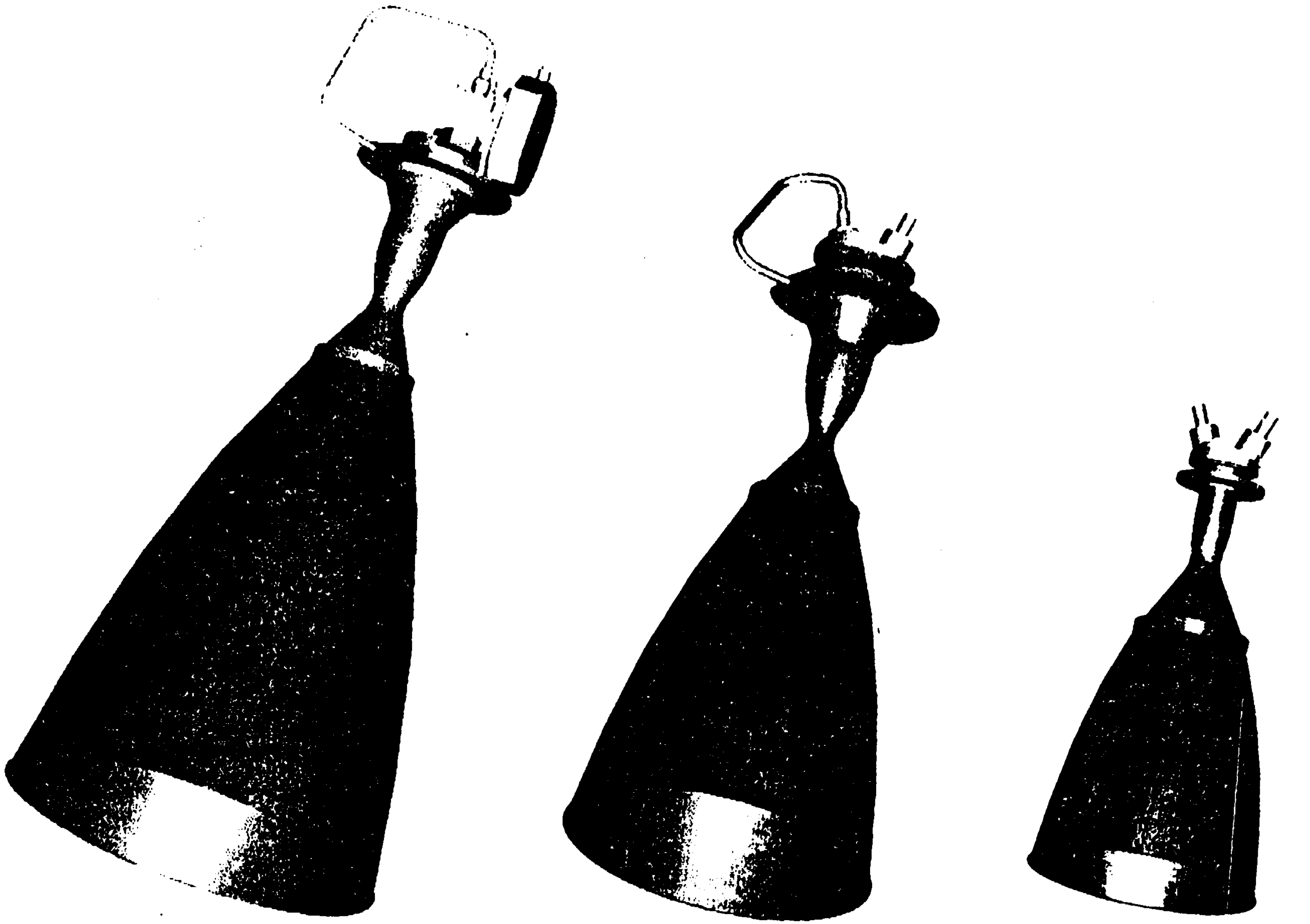
1.0 SUMMARY

We have studied the requirements for high pressure earth storable rocket technology. The potential applications and technologies have been identified, the appropriate ones for development are described, and the recommended plan for their development is given.

The rationale for the recommendations is given, along with data on recent propulsion experience, user-stated preferences, and recently active or potential commercial, DoD, and NASA programs which need the new technology. It is evident that unit cost is the selection parameter given highest weight by the user community.

To illustrate the expected results of this program, two conceptual designs of high-pressure rocket systems are given. One system is appropriate for existing pressure-fed propellant delivery systems with little or no modification to existing tankage or plumbing systems. The second, higher pressure system, would require changes to existing propellant delivery systems to be applied. The two conceptual rocket engines are shown in Figure 1-1. Both systems are derived from the demonstrated AJ10-221 Ir-Re 490 N engine, which is shown at the same scale. Both would use NTO/hydrazine at nominal thrust levels of 100 lbf. The thruster appropriate for existing propellant supply systems has a chamber pressure of about 250 psia; the pump-fed system has an operating chamber pressure of 500 psia. Other characteristics of the systems are summarized in Table 1-1, where they are compared to the baseline AJ10-221 engine, which was developed under NASA contract.

To prepare these conceptual designs, preliminary checks of performance, heat transfer, stability, stress, and cost were made to be sure none of these factors was violated. The basis for choice of thruster parameter space and for the high-pressure concepts presented are given in Section 3.0.



Reference - AJ10-221 Ir-Re NTO/MMH

P_c = 114 psia
F = 110 lbf
ε = 286:1
Is = 321.8 sec

Concept 1A - NTO/Hydrazine

P_c = 250 psia
F = 100 lbf
ε = 300:1
Is = 330 sec

Concept 2A - NTO/Hydrazine

P_c = 500 psia
F = 100 lbf
ε = 300:1
Is = 335 sec

Figure 1-1. Comparison of Flight Engine High Pc Concepts to Reference

CONCEPTUAL DESIGNS FOR HIGH-PRESSURE, EARTH-STORABLE ROCKET ENGINES

FLIGHT TYPE CONCEPT DESIGN=	AJ10-221 [REFERENCE]	#1A	#2A
DESIGN Pc, PSIA	115	250	500
THRUST, LBF	110	100	100
PROPELLANTS	NTO/MMH	NTO/HYDRAZINE	NTO/HYDRAZINE
Is, SEC	321.8	330	335
CHAMBER TEMPERATURE, oF	3380	3790	3950
ENVELOPE, MAX DIA., IN	13.8	9.2	6.5
ENVELOPE, MAX LENGTH	30	20.7	15
MAX WEIGHT, LBM	10	TBD	TBD
VALVE	MOOG TORQUEMOTOR	LOW COST	LOW COST
AREA RATIO	286	300	300
DESIGN LIFE, HOURS	>6	>12	>12
INJECTOR	S/N6-2	RE-BALANCE AJ10-221 FOR HYDRAZINE	USE BRILLIANT PEBBLES TI INJ.
THROAT DIA, IN	0.804	0.521	0.368
CHAMBER DIA, IN	1.71	1.71	0.57
CHAMBER MATERIAL	Ir-Re	LOW-COST Ir-Re	LOW-COST Ir-Re

Table 1-1

Conceptual Designs For High-Pressure,
Earth-Storable Rocket Engines

2.0 INTRODUCTION

This Task 1 informal report documents the selection of operating conditions for the High-Pressure, Earth-Storable Rocket Technology (HIPES) program. Factors considered included available or near-term advanced technology, user requirements, user acceptance and those applications with the most to gain from utilization of high operating pressure. This parameter space was evaluated to determine the "best" combination of propellant selection, thrust level, total impulse (operating time), and chamber material.

APPLICATIONS

Applications have been defined in terms of recent history and projects presently in the RFI, RFP, or early design selection stages. For example, in 1992 a total of 131 space vehicles were launched world-wide. Of these, 78 were launched by the C.I.S., who are not considered to be a potential market for our propulsion in the near term. Of the remaining 53 launches, the spacecraft were provided by U.S. companies in 19 launches; all are potential users of improved propulsion systems built in the U.S. Thirteen systems launched on Ariane are possible users. In discussions with ESA (Ref. 1) they indicated that U.S. companies will be considered as propulsion suppliers, although European companies are given preference.

Future applications for propulsion have been derived from space system models, user surveys, and user requests for information and quotations on specific propulsion applications. These sources indicate that the launch projections for the period through 2010 are similar to the actual experience for 1992. Over twenty near-term propulsion projects for new vehicles or upgrades to existing vehicles have been identified. These cover the rang from low orbit "light" satellites, to "heavy" communication satellites at GEO and large space-transfer "bus" propulsion. Base-lined propellants for these applications include NTO with either MMH or hydrazine, or ClF_5 and hydrazine. Thrust levels range (for axial as opposed to RCS engines) from 10 lbf to about 100 lbf. Propellant quantities range from about 150 lbm to 11,500 lbm. Most, but not all applications require obtaining the maximum propulsion system specific impulse that can be provided within the envelope constraints.

INCREASED CHAMBER PRESSURE

User surveys show that the advantages of higher chamber pressure (smaller envelope, higher performance) are appreciated. However, there is reluctance to move away from developed, qualified propellant delivery systems. The potential advantages of pump-fed systems are recognized by some users (in fact pump-fed 100 lbf engines were flown on Agena missions in the mid 1960's by LMSC); however, the overall user perception of pumps is that they are more expensive, less reliable, and, in some cases thought to be heavier than pressure-fed systems. Work is required to bring user acceptance of the ancillary systems required to achieve very high pressure.

COST CONSIDERATIONS

Both surveys of users and recent propulsion system competitions have shown that the primary discriminator used for system selection is cost. In fact, in the commercial market, low development and unit costs have more weight than demonstrated high performance. It becomes obvious that to meet the goal of user acceptance of these advanced propulsion technologies they must be cost competitive. The potential cost advantages to the users (greatly increased revenue due to longer life in orbit, for example) are diluted to obtain up-front returns. For example, changing from a conventional Cb chamber using NTO/MMH at 100 psia Pc to an Ir-Re chamber using NTO/hydrazine at 250 psia Pc could result in a nearly 40% increase in revenue over the extended life of the satellite, or in $2M up-front return if taken as off-loaded propellants. Since the latter return is realized whether the launch is successful or not, typically it is selected, reducing the long term profit potential for the advanced propulsion.

Because of the premium on short term profit, the amount of added investment that can be made for higher performance thrusters is limited. For this reason, both non-recurring and unit costs are critical. Basic considerations show that Ir-Re chambers will always be more expensive than Cb chambers, for example; some cost differential can be sustained and still remain a viable alternative. The present cost differential must be reduced, however, to become competitive. The fabrication development work for Ir-Re being funded by NASA LeRC should improve this position.

3.0 CANDIDATE SYSTEM DEFINITION

An earth-storable, high-performance small thruster has many potential applications, from short-pulse divert engines to multi-hour delta V applications. We have studied the potential applications from several different view points: 1) What is the very recent history for application of this class of thruster?, 2) What do the users and mission planners as a group determine to be their on-going and future needs?, and 3) What are real applications based on Requests for Information (RFI) and Requests for Proposal (RFP) for these systems?

3.1 POTENTIAL APPLICATIONS

The answers to these questions have been used to define the applications space for small, earth-storable high performance thrusters, within the stated guidelines of this program. The results of this study show that the highest value applications, with the most chance of user acceptance and utilization, are for large delta V propulsion, in the 100 lbf thrust class, with NTO/hydrazine as propellants. The data which lead us to these conclusions are provided below, along with two flight engine design concepts which fit these conclusions.

3.1.1 Historical Applications Basis

To focus on actual propulsion applications, the space launches conducted in 1992 were reviewed. Table 3.1.1-1 summarizes these launches by agency. There were a total of 131 space launches in 1992; however, 78 of these were conducted by the C.I.S. Of the remaining launches, 12 were by NASA, 12 by DoD, 5 were U.S. commercial, and 24 were foreign. Of these foreign launches, 14 had payload/propulsion systems provided by U.S. spacecraft manufacturers. These spacecraft used conventional thruster technology: Cb chambers, "low" Pc, and, if bipropellant, NTO/MMH.

Table 3.1.1-1 also shows projected launches per year for the categories of interest, through the year 2010, based on the draft and final Mission Model Study prepared by the NSIA Spacecraft Panel (Refs. 3 and 4). The difficulty of predicting the future, and the danger of relying on such predictions, is obvious. Nevertheless, the indication from this study is that the average yearly spacecraft launch rate will be similar to actual experience for 1992, with a reduction projected for foreign launches. However, because of overwhelming cost considerations, it appears more likely that more will be on lower cost foreign launchers (Ariane, Long March, and perhaps even on C.I.S. launchers).

The 1992 launches (except C.I.S.) are listed in Table 3.1.1-2, which shows spacecraft manufacturer, spacecraft, application, spacecraft launch weight, and launch vehicle.

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SYSTEM SELECTION PARAMETER--
USER AGENCIES AND LAUNCH FREQUENCY

AGENCY		TOTAL LAUNCHES 1992 [1]	PROJECTED LAUNCHES/YR 1992-2010 [2]
NASA		12	6+SHUTTLE
DoD		12	11
COMMERCIAL, US		5	4
FOREIGN (EXCEPT C.I.S.)		24	10
C.I.S.		78	N/A
	TOTAL=	131	31+SHUTTLE+C.I.S.

[1] TRW SPACE LOG

[2] SOURCE: MISSION MODEL SUMMARY, NSIA SPACECRAFT PANEL, [DRAFT], 7-23-92 AND FINAL REPORT, 4-30-93

Table 3.1.1-1

System Selection Parameter – User Agencies and Launch Frequency

The average weight at launch for these 35 spacecraft was just over 2000 lbm, suggesting an average propellant load of about 1600 lbm, of which approximately 1300 lbm would be available for orbit transfer and the balance for on-orbit station-keeping/attitude control. These data provide guidance for required thrust levels and total durations which, along with practical limits on safe total firing time, aid in selection of thrust level required.

Conventional Technology

Base-line conventional technology for small earth-storable thrusters is represented by silicide-coated Cb chambers operating with NTO/MMH propellants at Is from 285 to 310 depending on thrust level. A summary of typical thrusters for eight propulsion system manufacturers, both U.S. and foreign is shown in Table 3.1.1-2. This table also shows data for some advanced thrusters which are under development, to give an indication of some of the present directions being taken to obtain higher performance.

Advanced Technology

The status of advanced technology thruster development, as exemplified by the Ir-Re chamber technology at Aerojet is shown in Table 3.1.1-3. Others known to be pursuing this technology recently are TRW and Royal Ordnance. Based on Aerojet's experience, with proper design this material system works well and has demonstrated over 15 hours duration at the 5 lbf thrust level and over 6 hours at the 110 lbf thrust level, at conventional chamber pressures, with NTO/MMH propellants. Neither of these durations represent upper limits. The advantage of this material system over Cb is that no film cooling is required, permitting higher performance for a given propellant and chamber pressure. An alternate approach is being explored in the U.S. and Russia to develop a higher temperature barrier coating to replace the slicide coating now used. To match the Ir-Re performance these chambers must be operated at about 3300°F.

3.1.2 User Survey

A survey of users of small thrusters was conducted in Spring 1993. Forty-six positive contacts were made with 42 propulsion groups. A total of 63 surveys were sent out; 22 responses were received, a good return for this type of survey. The survey questions and their answers are given in Table 3.1.2-1, Mission Data, Table 3.1.2-2, Pressurization Systems, and Table 3.1.2-3, Thruster Systems.

SMALL EARTH-STORABLE LIQUID ROCKET ENGINES FOR SATELLITE AND SPACE VEHICLES

SUPPLIER	MODEL	THRUST	PROPELLANTS	CHAMBER MATERIAL	BURN TIME	Pc	MR	Is	AREA RATIO	MASS	DATA SOURCE
AEROJET		2.00N	NTO/MMH				1.65	285	150		INTER. SPACE. DIR/90-91
AEROJET		21.35N	NTO/MMH				1.65	285	150		INTER. SPACE. DIR/90-91
AEROJET		445N	NTO/MMH				1.65	309	150		INTER. SPACE. DIR/90-91
ATLANTIC RSEARCH		22-4500N FAMILY		COLUMBIU							INTER. SPACE. DIR/90-91
DEUTSCHE AEROSPACE		4N	NTO/MMH	Pt-Rh	>40 HRS	7 BAR	1.64	285	187	270 gm	AIAA 93-2120
DEUTSCHE AEROSPACE		10N	NTO/MMH	Pt/Rh	>40 HRS	9 BAR	1.65	290	150	300gm	AIAA 93-2120
DEUTSCHE AEROSPACE		400N				10 BAR	1.643	317	220	3117kg[elc	AIAA 93-2120
MARQUARDT	R-1E	110N	NTO/MMH	Cb	82000		1.65	280	100		INTER. SPACE. DIR/90-91
MARQUARDT	R-40A	3870N	NTO/MMH	Cb	500 MAX/15,319	10.5 ATM	1.6	281/306	20/120	10.25KG	INTER. SPACE. DIR/90-91
MARQUARDT	R-4D	490N	NTO/MMH or AH	Cb	UP TO 1 HR	6.84 ATM	1.65	312		3.63 KG	INTER. SPACE. DIR/90-91
MARQUARDT	R-42SR	140-300LBF	NTO/MMH		>3E6LBF-SEC	350-150		303	164		AIAA 93-2118
MARQUARDT	R-6C	22N	NTO/MMH	Cb	UNLIMITED		1.6+-.1	290	100	0.66 KG	
MARQUARDT	R-6C-2.2	10N									INTER. SPACE. DIR/90-91
MARQUARDT	R4-D	490N									AIAA-93-2517
ROCKET RESEARCH CO.	MR-50, -103, -104,		]-HYDRAZINE MONO.								INTER. SPACE. DIR/90-91
ROCKET RESEARCH CO.	-107, -111, -501		-501 IS ELEC. AUG.					280-304(EP)			INTER. SPACE. DIR/90-91
ROYAL ORDNANCE	LEROS-1	105LBF	NTO/HYDRAZINE	Cb,TI INJ	22.8Ksec(1990)	80-100	0.7-1.0	318(320 '93)	200		PRESENT. AT APD, 5-28-93
ROYAL ORDNANCE	LEROS-2	125LBF	NTO/MMH					311			PRESENT. AT APD, 5-28-93
ROYAL ORDNANCE	LEROS-2A		NTO/MMH	"ADV. MATL				320('93)	300		PRESENT. AT APD, 5-28-93
ROYAL ORDNANCE	LEROS-20	5LBF	NTO/MMH	Cb				295			PRESENT. AT APD, 5-28-93
ROYAL ORDNANCE	LEROS-20A	5LBF	NTO/MMH	"NEW MATL				295('92)310('			PRESENT. AT APD, 5-28-93
ROYAL ORDNANCE	LEROS-20H	NTO/HYDRAZINE	NTO/HYDRAZINE					TARGET 300(			PRESENT. AT APD, 5-28-93
ROYAL ORDNANCE	LEROS-20HA[?]	5LBF	NTO/HYDRAZINE	ADV. MATL				TARGET 313			PRESENT. AT APD, 5-28-93
TRW	DUAL-MODE		NTO/HYDRAZINE								INTER. SPACE. DIR/90-91
TRW	ERIS										INTER. SPACE. DIR/90-91
TRW	MRE-0.1,-1, -4		]-HYDRAZINE MONO.								INTER. SPACE. DIR/90-91
TRW	-R/GRO, -15/OMV, -50		]								INTER. SPACE. DIR/90-91
TRW	OMV	VARI.: 57.8-578N	NTO/MMH	Cb ALLOY		0.75-6.8 AT	1.64	280-308	125	6.8 kg	INTER. SPACE. DIR/90-91
UNITED TECH/HAM STD	REA 10, 17-6,	]-.89-26.7N	]-HYDRAZINE MONO.								INTER. SPACE. DIR/90-91
UNITED TECH/HAM STD	17-12, -16, 39-2	]	]								INTER. SPACE. DIR/90-91

Table 3.1.1-2

Small Earth-Storable Liquid Rocket Engines for Satellite and Space Vehicles

Table 3.1.1-3

Aerojet's Ir/Re Chambers Have Demonstrated Nearly 50 Hours of Hot Firing

Thrust, lbf	Area Ratio	Max Temp, °F	Max O/F	Starts	Full Thermal Cycles	Test Duration, sec
5	8.4:1	4,200	2.1	3,638	37	31,369
5	8.4:1	4,100	1.7	14	14	13,016
5	8.4:1	4,300	1.7	157	74	28,426
5	8.4:1	4,070	2.0	2,701	70*	>54,431
5	8.4:1	3,920	1.4	10	9*	>926
5	150:1	4,000	1.9	>94,588	32*	>4,788
5	150:1	3,607	1.7	>100,000	28	7,735
14	75:1	3,553	1.9	306	19*	>314
14	75:1	Not Tested	Not Tested	Not Tested	Not Tested	Not Tested
110	22:1/44:1	3,500	1.7	47	45*	>3,884
110	22:1/44:1	3,500	1.7	60	57*	>16,728
110	286:1	3,391	1.8	89	77*	14,076
110	286:1 Cutback to 47:1	3,600	1.65	4	3	8,499
110	286:1	Not Tested				
Totals				>201,610	462	>184,192

* Chambers show no evidence of coating loss or cracking due to fatigue. Ultimate life capability not determined yet. Total firing time = 175,693 sec (48.8 hr).

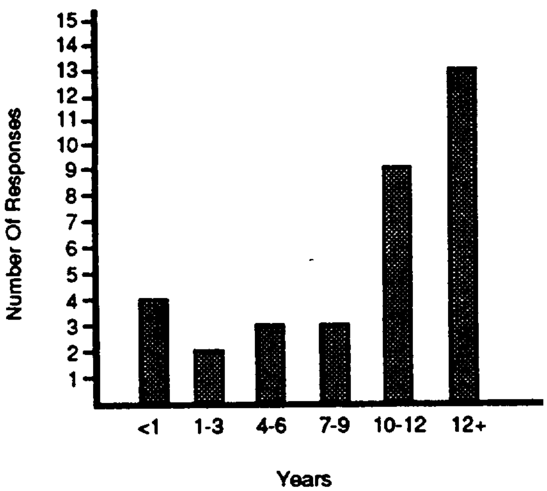
June, 1993

Mission Data

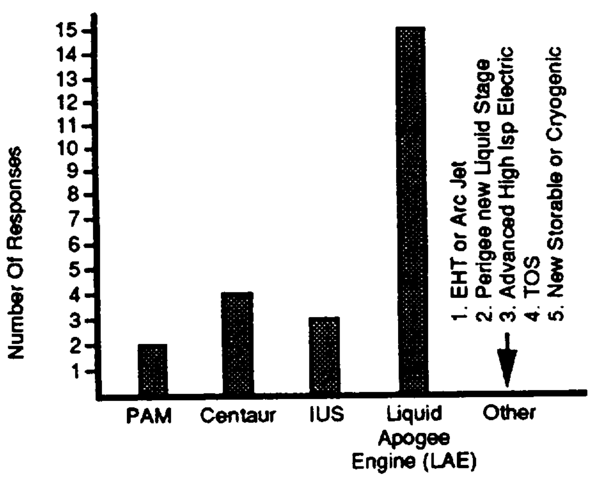
Survey Results

Mission Data

1. What are your required satellite propulsion system on-station operability times:



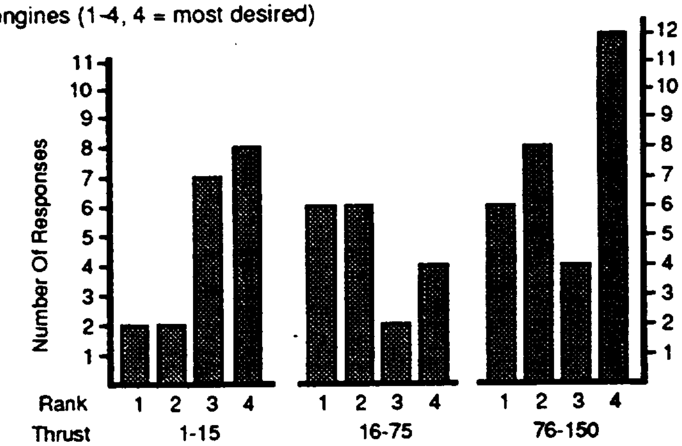
If you integrate apogee delta V into your satellite, how would you accomplish GTO transfer and GEO circularization:



3. Please rank the total impulse per satellite requirement you anticipate in the near future:

note: the question was ambiguous and results therefore inconclusive

4. Please rank in order of unsatisfied need for desired engines (1-4, 4 = most desired)

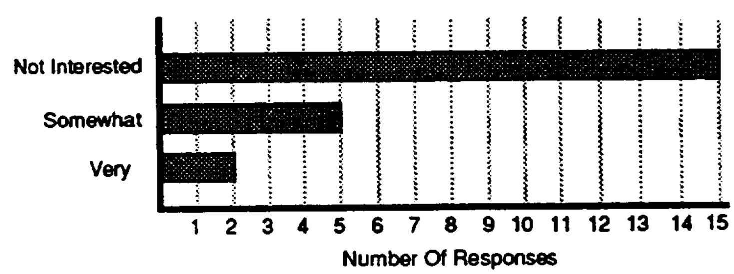


What is your preferred thrust level

Axial Engine:	90 - 110	(1)
	100 - 110	(7)
	100 - 200	(4)
	1000+	(1)
Reaction Control Thrusters:	≤1	(6)
	2-5	(9)
	>5	(2)

What is your maximum acceptable satellite g level:
≤.1 (9) .2 - 1 (4) >1 (4)

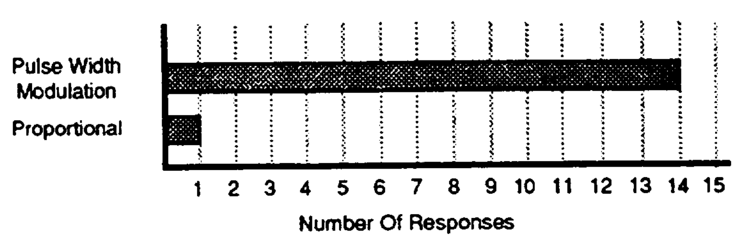
5. Is throttlability of interest:



6. What is your minimum impulse bit (lbF-sec):

< .03 (4) .03 - .05 (6) >.05 (2)

7. Which is preferred:



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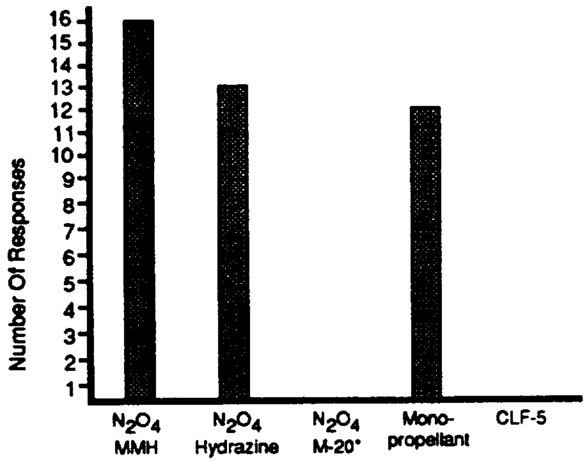
Mission Data (cont.)

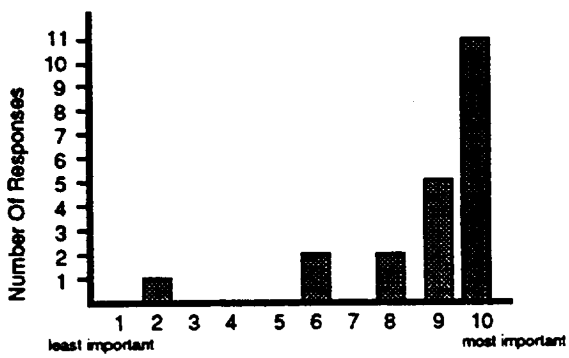
8. What is your preferred propellant combination:



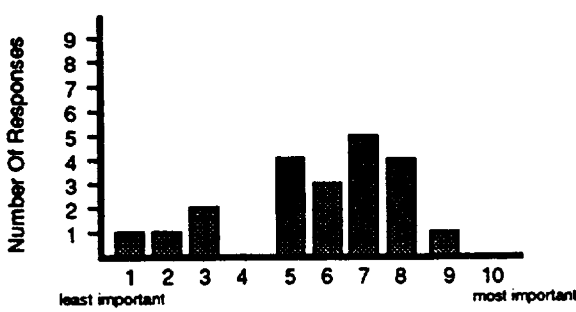
* M-20 = 80% Hydrazine + 20% MMH

(Note: H_2O_2 / RP-1 of interest to 1 respondent)

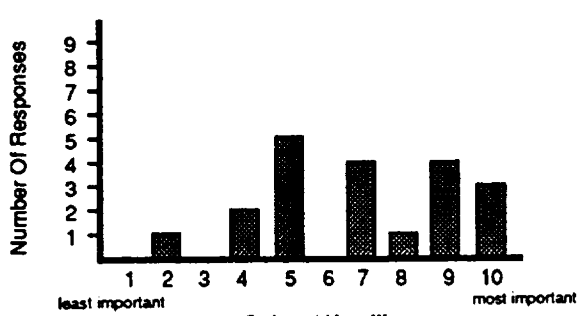
9. Please provide a weighting factor (1-10, 10 = most important) for the following propellant parameters:



Performance (I_{sp})

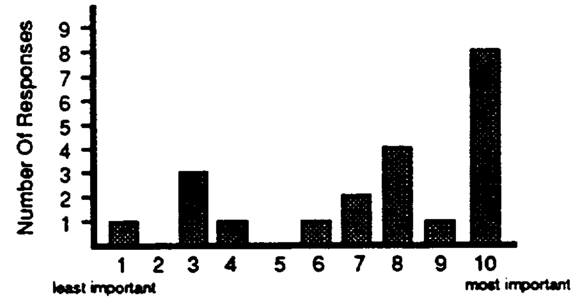


Volume (Density)

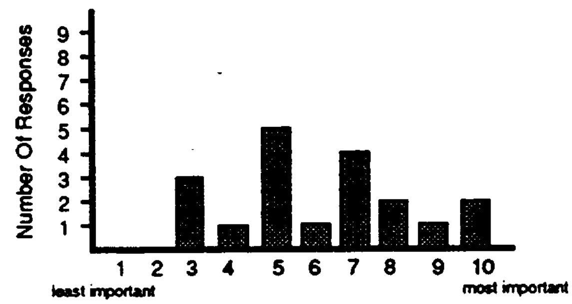


Safety / Handling

(9. cont.) Please provide a weighting factor (1-10, 10 = most important) for the following propellant parameters:



Cost



Storage Difficulty

Table 3.1.2-2

Pressurization Systems

Pressurization Systems

1. What is your pressurant for pressure fed systems:

Helium: yes (21 responses)

Other: GN_2 (5 responses)

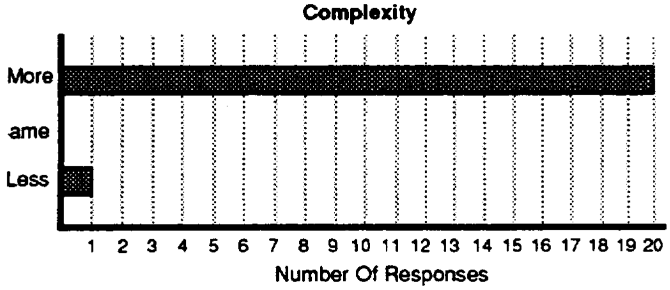
2. Have you ever considered a pump fed pressurization system for you satellite as a means of reducing weight?

Yes : (9 responses)

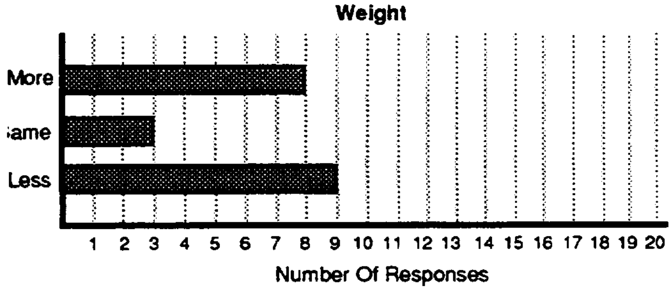
No : (11 responses)

3. What is your perception of an on-board pump fed liquid propulsion system for orbit raising and circularization as compared to a pressure fed system?

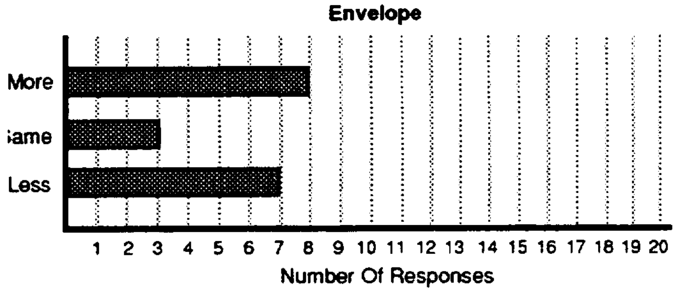
Complexity



Weight

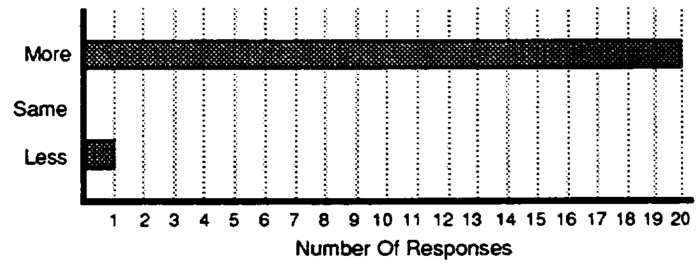


Envelope

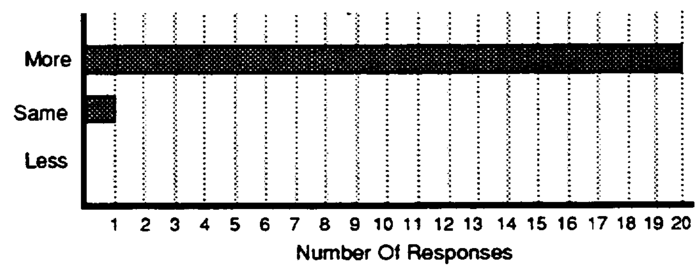


(3. cont.) What is your perception of an on-board pump fed liquid propulsion system for orbit raising and circularization as compared to a pressure fed system?

Risk / Reliability

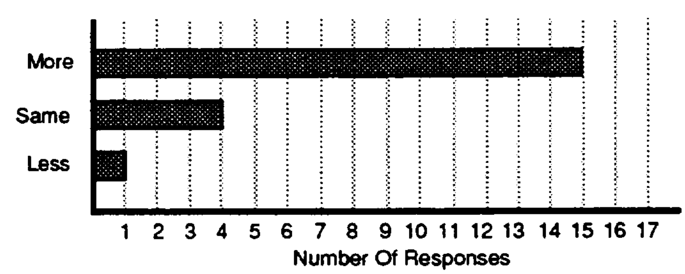


Cost

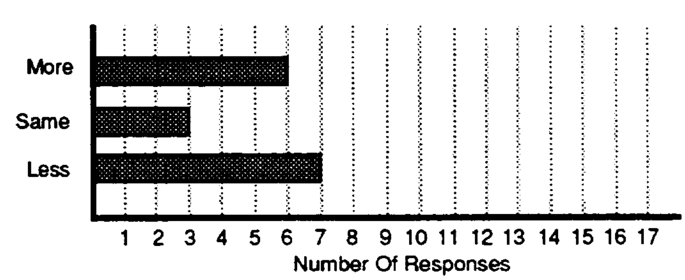


4. What is your perception of a pump-fed LAE as compared with a pressure fed system?

Complexity



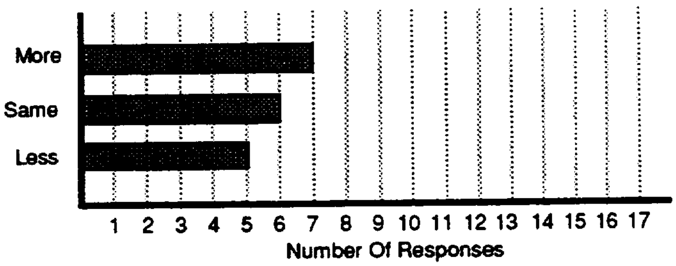
Weight



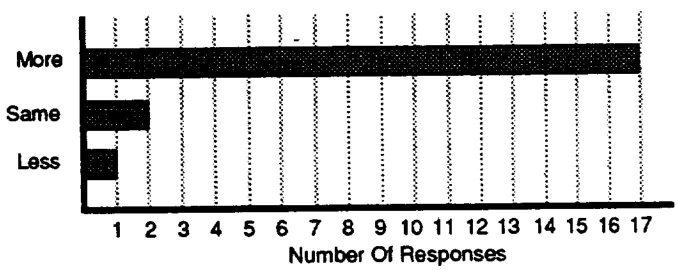
Pressurization Systems (cont.)

(4. cont.) What is your perception of a pump-fed LAE as compared with a pressure fed system?

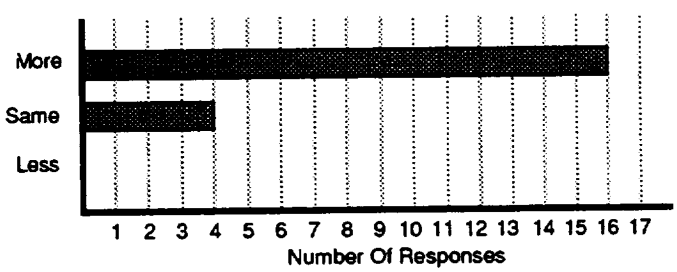
Envelope



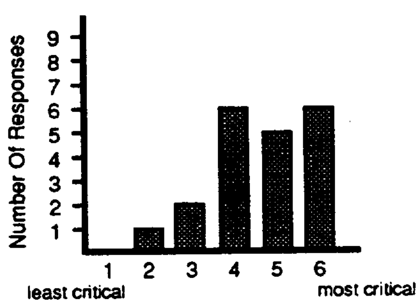
Risk / Reliability



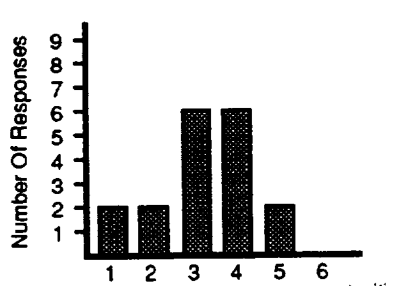
Cost



5. Please rank in order of criticality (1-6, 6 = most critical) the factors against your use of a pump fed LAE:

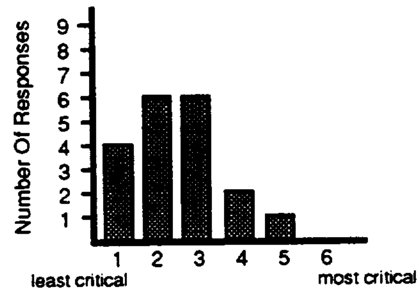


Complexity

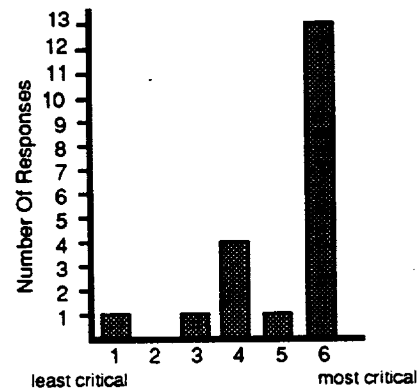


Weight

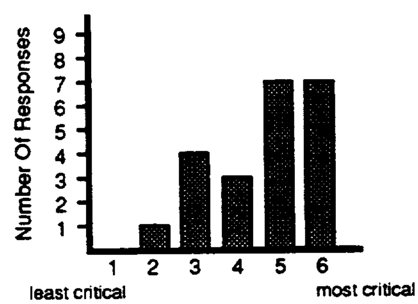
(5. cont.) Please rank in order of criticality (1-6, 6 = most critical) the factors against your use of a pump fed LAE:



Envelope



Risk / Reliability



Cost

6. Based on the assumption that a pump fed LAE could be qualified for flight in accord with your specifications and includes a 20 second I_{sp} increase over conventional chambers (I_{sp} = 310 sec):

Would you develop an on-board propulsion system to use this LAE:

Yes : (11 responses)

No : (9 responses)

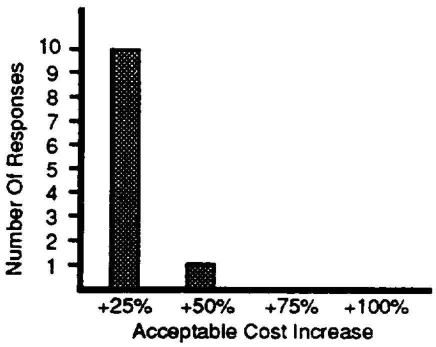
Would you buy a complete system for integration with your stage?

Yes : (9 responses)

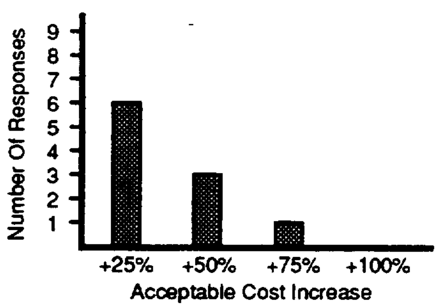
No : (10 responses)

Pressurization Systems (cont.)

7. How much would you be willing to pay for the LAE or the system as compared to existing designs?

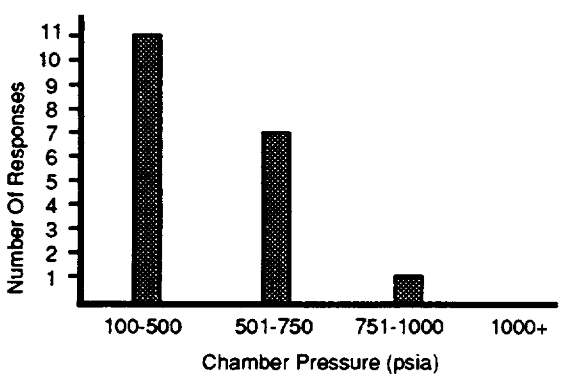


LAE with I_{sp} = 330 sec



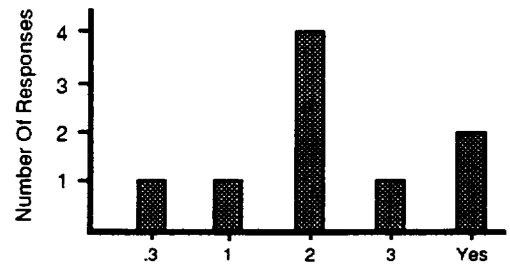
LAE with 10% lower dry weight and I_{sp} = 330 sec

8. At what minimum chamber pressure (psia) would you consider pump use:

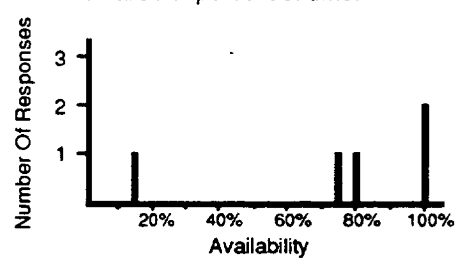


9. What sources of energy are available to pressurize the propulsion system

Solar Cell Electrical (kW):



Available percent of time:

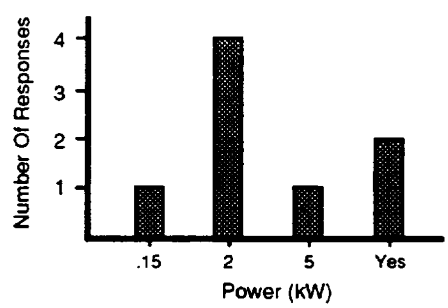


Available during LEO to GEO transfer:

Yes (8 responses)

No (3 responses)

On Board Batteries (kW):



(kW-hr): 2 (1 response)

recharge time: 6, 8, 12 hr (3 responses)

Available during LEO to GEO transfer:

Yes (6 responses)

No (2 responses)

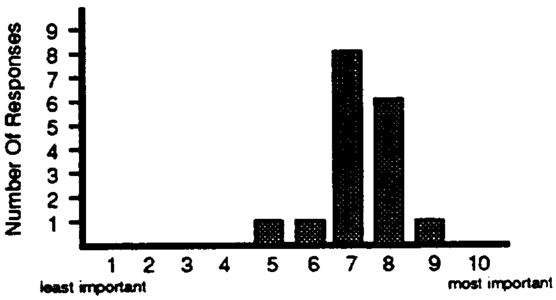
Are there any other energy sources available?

Yes RTG's (1 response)
Pneumatic GHe (1 response)

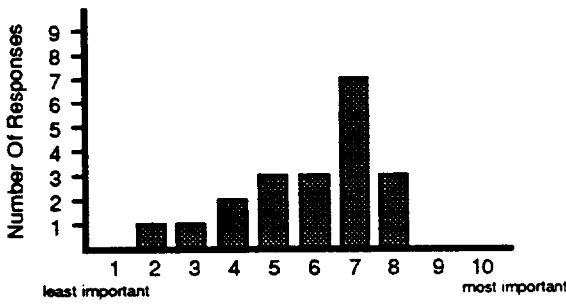
No (10 responses)

Thruster Systems

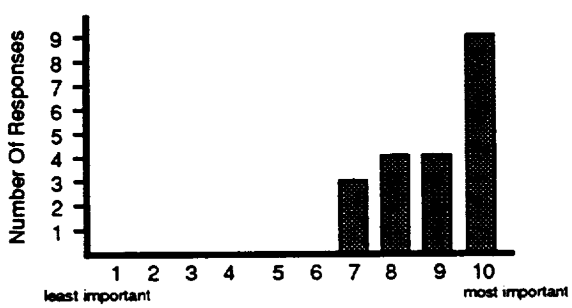
1. Please provide a comparative weighting factor 1-10 (10 = most important) for the following thruster attributes:



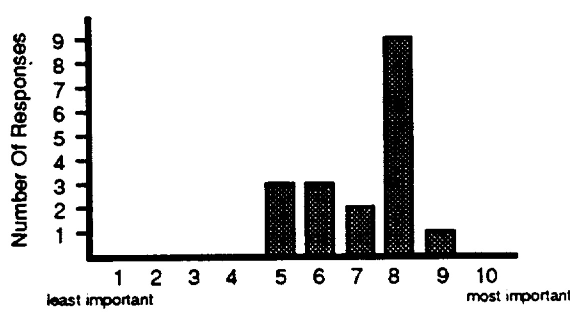
Weight



Packaging Envelope

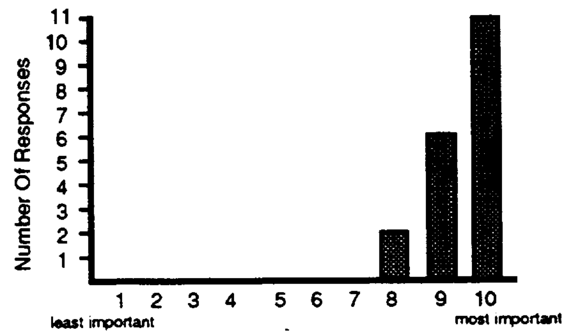


Cost



Thrust

(1. cont.) Please provide a comparative weighting factor 1-10 (10 = most important) for the following thruster attributes:



I_{sp}

2. Please rank the importance and number of satellite thruster used per satellite:

	Rank	#
Orbit Raising / Insertion (LAE)	inconclusive data	
Planetary Transfer / Retro		
Station Keeping / Reaction Control (RCT)		

3. Do you consider plume impingement to be a major concern?

   Yes : (11 responses)

   No : (5 responses)

4. Can thrusters be buried if satellite components are shielded from radiant heat?

   Yes : (12 responses)

   No : (6 responses)

5. Please provide (if possible) the satellite partials for the following:

   Propellant Partial (# payload / # propellant)

   .1, .21, .3, .5, .7, .8, .9, 1.2, 2.2, 6 (10 responses)

   I_{sp} Partial (#payload / sec I_{sp})

   .3, .4, 2.4, 5 (2), 8.5, 10, 11 (8 responses)

6. How much, if any, heat conduction is acceptable from a given engine through it's mounting structure:

   None : (5 responses)

   Watts : 50 (1 response)

   BTU/hr : some (1 response)

The overall results can be summarized as described below.

MISSION DATA

1. Operating time: 10 to 12 years
2. Delta V method: liquid engine
3. Total impulse required: [ambiguous wording; can not interpret]
4. Thrust level desired: 100 to 150 lbf
5. Is throttlability required? No
6. Minimum impulse bit? [ambiguous – answered only for lowest thrust]
7. Pulse-width or proportional control? Pulse-width
8. Preferred propellants? NTO/MMH 39%; 32% NTO/Hydrazine; 29% mono-prop
9. Provide weighted preference for the following:
   - performance: 88% of possible maximum score; ranking = 1.00
   - Cost: 74% of possible maximum score; ranking = 0.84
   - Safety: 69% of possible maximum score; ranking = 0.78
   - Storage ease: 62% of possible maximum score; ranking = 0.70
   - Volume: 59% of possible maximum score; ranking = 0.66

PRESSURIZATION SYSTEMS

1. Pressurant? Helium

2. Considered pump-fed? 45% have considered

3. Pump fed propellant system perceptions:

   - Complexity: 75% believe more complex than pressure-fed
   - Weight: about even split on heavier/lighter
   - Envelope: about even split on larger/smaller
   - Risk/reliability: 95% favor pressure-fed
   - Cost: 95% favor pressure-fed

4. Pump-fed liquid engine perceptions:

   - Complexity: 75% believe more complex than pressure-fed
   - Weight: about even split on heavier/lighter
   - Envelope: about even split on larger/smaller

- Risk/reliability: 94% favor pressure-fed
- Cost: 80% favor pressure-fed

5. Rank factors against use of pump-fed thrusters:

    - Risk: 86% of possible maximum score; ranking = 1.00
    - Cost: 78% of possible maximum score; ranking = 0.91
    - Complexity: 78% of possible maximum score; ranking = 0.90
    - Weight: 54% of possible maximum score; ranking = 0.63
    - Envelope: 41% of possible maximum score; ranking = 0.48

6. If pump-fed specific impulse is 20 sec higher than conventional, would you:

    - Develop a system? 55% yes
    - Buy a system for integration? 47% yes

7. If LAE has Is = 330 sec:

    - Acceptable cost increase? + 25% cost increase acceptable

    - Higher performance and 10% lower dry weight: 25% to 50% cost increase acceptable.

8. What is minimum Pc you would consider for pump-use?
    - Generally in the range of 500 psia

9. Energy sources for pump:
    - Solar cells, 1 to 3 kw, available 70 to 100% of time, and, (73% of responses) available during LEO/GEO transfer
    - Batteries, 2 to 5 kw, 2 kW-hr, 6 to 12 hr recharge, 75% have available during transfer
    - Other power sources? None

THRUSTER SYSTEMS

1. Comparative ratings:

    - Specific impulse: 95% of possible maximum score; rank = 1.00
    - Cost: 90% of possible maximum score; ranking = 0.94

- Weight: 73% of possible maximum score; ranking = 0.77
- Thrust: 71% of possible maximum score; ranking = 0.75
- Packaging: 60% of possible maximum score; ranking = 0.63

2. Rank thrusters by importance and quantity/satellite: inconclusive responses.

3. Is plume impingement a major concern? Yes

4. Can thrusters be buried if thermally shielded? Yes

5. provide payload mass trades for:
   - Propellant mass to payload mass trade-off. Illogical response – question not clear.
   - Specific impulse to payload mass trade-off. Illogical responses

6. Allowable heat conduction from thruster? None

3.1.3 Current Applications Based on Recent Customer RFI/RFP

Active or recently active procurements and potential procurements in the small earth-storable thruster area within our experience have been reviewed. These programs, which represent real or potential thruster applications, are listed in Table 3.1.3-1.

Procurements in several of these programs have been or are being finalized; the balance are in various stages from initial discussions to formal proposal.

The table shows that there are a predominance of 100 lbf-class requirements, several of which would use NTO/Hydrazine. Three applications have baselined ClF_5/AH as propellants to provide what appears to be the highest performance available in an earth-storable propellant combination. At present these are considered to be applications where the propellant is enabling but the total number of spacecraft is small, and significant technology development is required, both for infra-structure and long-life chamber materials.

The transitory nature of some of these applications is typified by the Bus-1 (LMSC/JFC); this would be used on the Option A Space Station whose design was announced in late May '93 (Ref. 5). The selection between Options A, B and C is scheduled to be announced as this report is being prepared, although it appears that the propulsion system decision will be finalized later.

Table 3.1.3-1

Recent Small Earth Storable
Thruster Active Procurements

HIPC95
8-9-93

RECENT SMALL EARTH STORABLE THRUSTER ACTIVE PROCUREMENTS

UPDATE 9-13-93

PROGRAM	CUSTOMER	ENGINE SIZE	CHAMBER MATERIAL	PROPELL.	AEROJET POSITION	START DATE
ORACLE--DESCENT	CARNEGIE-MELLON		HfC	CIF5/AH	BASELINED	
ORACLE--TLI	CARNEGIE-MELLON		HfC	CIF5/AH	BASELINED	
GBI	SDIO	5 TO 20	SIC	NTO/MMH	ROCKETDYNE	ON-GOING
FEWS	LMSC	5	Cb	NTO/MMH	BASELINED	94/95
MILSTAR	LMSC	5	Cb	NTO/MMH	REPLACEMENT	1994
MISTI	PHILLIPS LAB	5		NTO/MMH	PLANNING	
NORSTAR	E PRIME	5	Cb	NTO/MMH	BASELINED	1994
IRIDIUM	LMSC	10	Cb?	AH (MONO)	FALL-BACK	1993
BUS-1	LMSC/JFC	14	Ir-Re	NTO/MMH	UPGRADE	TBD
BUS-1	LMSC/JFC	14	Cb	NTO/MMH	BASELINED	94/95
CLASSIFIED	LMSC	14	Ir-Re	NTO/MMH	BASELINED	1994
THAADS	SDIO	20	SIC	NTO/MMH	BASELINED	ON-GOING
ARTIMUS	JSC	100	Cb	NTO/MMH	MARQUARDT	1994
BRILLIANT PEBBLES	BMDO	100	SIC	NTO/MMH	BASELINED	ON-GOING
CLEMENTINE 1	BMDO	100	Cb	NTO/MMH	MARQUARDT	ON-GOING
LEAP	BMDO	100	SIC	NTO/MMH	RODKETDYNE/MARQU.	ON-GOING
LESSR	MSC	100	SIC	NTO/MMH	ROCKETDYNE	1993
MESUR	JPL	100	Cb	NTO/MMH	MARQUARDT	1994
ACAT	PHILLIPS LAB	100 TO 5000	HfC	CIF5/AH	BASELINED	94/95
BUS-1	LMSC/JFC	110	Ir-Re	NTO/MMH	UPGRADE	TBD
CLASSIFIED	LMSC	110	Ir-Re	NTO/MMH	BASELINED	1994
EROS EXPLORER	JOHNS HOPKINS JPL	110	Ir-Re	NTO/AH	INITIAL DISCUS.	
FEWS	LMSC	110	Ir-Re	NTO/MMH	BASELINED	94/95
GE(MARTIN) BUS	GE	110	TBD	NTO/AH	TRW/FALL-BACK	1992
H-601 BUS	HUGHES	110	TBD	NTO/MMH	MARQU/FALL-BACK	1993
METOP	BRIT. AEROSP.	110	Ir-Re	NTO/AH	INITIAL PLAN.	
NORSTAR	E PRIME	110	Ir-Re	NTO/MMH	BASELINED	1994

We see a trend towards NTO/Hydrazine in an effort to obtain higher performance than NTO/MMH with minimum change in spacecraft overall design. The driver here is the spacecraft manufacturer's ability to provide either more capability or lower cost to the spacecraft users to remain competitive.

3.2 SYSTEM PARAMETER SELECTION

The previous Sections have provided the application/user data which we have used to define the propulsion system parameters out of the wide range of parameter space considered.

As one bound of the problem, the RFP/contract specifies possible and excluded parameter ranges. As a reference, these parameters and their limitations are summarized in Table 3.2-1.

A large degree of flexibility is permitted; the "hard" constraints are: 'earth-storable', 'user acceptance and frequency of occurrence', and not 'divert or attitude control propulsion'.

The results of the parameter selection are summarized in Table 3.2-2; the bases for the choices are discussed in the following Sections, beginning with an overall review of spacecraft optimization, followed by selection of operating pressure, propellants, thrust, and total impulse. Included is discussion of the cost impacts which must be considered to assure a viable propulsion system which will actually be accepted by spacecraft manufacturers.

3.2.1 Systems Optimization

Selection of the specific operation points in the multi-parameter space which defines the engine system requires an understanding of its effect on the spacecraft system. We have conducted system trade studies which show the effect of chamber pressure and spacecraft pressurization system design on spacecraft performance. These system studies use our ELES computer code for propulsion system design and optimization. The engine performance parameters used in the code are given in Section 3.3.1.

We determined the weights of pressure-fed and pump-fed propulsion systems as a function of chamber pressure for large and light-sat applications. The comparison for a large spacecraft (8000 lbm launch weight) is shown in Figure 3.2.1-1. The mass of the pump-fed system is very nearly constant over the Pc range to 500 psia, while the pressure-fed system shows a nearly linear increase in mass.

Table 3.2-1

System Parameter Limits Specified in RFP for Selection in Task 1 of High Pc Propulsion Technology

HIPC54

8-2-93

SYSTEM PARAMETER LIMITS SPECIFIED IN RFP FOR SELECTION IN TASK 1 OF HIGH Pc PROPULSION TECHNOLOGY

SELECTED PARAMETER	RFP LIMIT/DEFINITION
1. POSSIBLE USER AGENCIES	NASA, DoD, CIVIL SPACECRAFT
2. CHAMBER PRESSURE	HIGH
3. STORABILITY	EARTH STORABLE
4. PROPELLANTS TO CONSIDER	NTO/MMH, NTO/AH, OTHERS
5. POTENTIAL APPLICATIONS	APOGEE INSERTION, DELTA V, PLANETARY RETRO
6. EXCLUDED APPLICATIONS	DIVERT OR ATTITUDE CONTROL PROPULSION
7. VEHICLES TO CONSIDER	"LIGHT" AND "HEAVY" SATELLITES AND SPACECRAFT
8. PRESSURIZATION SYSTEM	PRESSURE-FED AND PUMP-FED
9. OTHER FACTORS TO CONSIDER	EVENTUAL USER ACCEPTANCE EXPECTED FREQUENCY OF OCCURENCE
10. SELECT PREFERRED ENVELOPES FOR	THRUST, CHAMBER PRESSURE, MR, ETC.
11. TECHNOLOGY DEVELOPMENT INCLUDES	INJECTOR CHAMBER NOZZLE APPLICABLE ADVANCED MATERIALS SUCH AS, BUT NOT LIMITED TO, Ir-Re
12. TECHNOLOGY DEVELOPMENT EXCLUDED	VALVES OR PUMPS
13. INVESTIGATE	COMBUSTION EFFICIENCY AND HEAT TRANSFER AT MIN. OF 3 Pc, AT FIXED THRUST

ALL ITEMS LOCATED IN RFP PG J-5, TASK 1

HIPC PARAMETER SPACE
BASES FOR PARAMETER SELECTION

PARAMETER	SELECTION	BASIS
COST	LOW COST DESIGN APPROACH: [TO BE DETERMINED]	SHORT TIME FRAME FOR INVESTMENT PAY-BACK FORCES LOW UP-FRONT COSTS AND LOW UNIT COSTS: POTENTIAL SAVINGS OF LONGER SATELLITE OPERATING LIFE OCCUR TOO LATE [A DECADE OR MORE AFTER LAUNCH]; MUST COMPETE ON COST TO BE A VIABLE APPLICATION
PROPELLANTS	N204/N2H4	INCREASED PERFORMANCE OVER NTO/MMH USER TREND IS TOWARDS NTO/AH SURVEY SHOWS NEARLY EQUAL WEIGHT FOR MMH & AH (16 vs. 13) PERMITS DUAL MODE OPERATION NEXT GENERATION; CONSISTANT WITH PROGRAM TIME SCALE PROVIDES TRANSITION TO ELECTRIC PROPULSION PROVIDES MARKETABLE NEW TECHNOLOGY FOR APD NO USERS SURVEYED EXPRESSED INTEREST IN CIF5 ONLY POTENTIAL CIF5 APPLICATIONS HAVE SMALL EXPECTED FREQUENCY OF OCCURENCE INFRASTRUCTURE FOR CIF5 NOT CONSISTENT WITH PROGRAM REQUIREMENTS
THRUST LEVEL	100LBF-CLASS	AXIAL ENGINE HAS BIGGEST PAYOFF THRU PROPELLANT SAVINGS TYPICALLY 80% OR MORE OF PROPELLANT MASS DIVERT OR ATTITUDE CONTROL PROPULSION SPECIFICALLY EXCLUDED IN CONTRACT TREND IS DOWNWARDS IN THRUST (e.g. FROM 200 TO 100LBF) SURVEY SHOWS 8/5 OR BETTER FOR 90-110 LBF CLASS
Is GOAL	335 SEC	REQUIRED TO BE COMPETETIVE
TOTAL IMPULSE	3,000,000LBF-SEC MINIMUM	EQUIVALENT TO 8+ HOURS OF FIRING TIME AT 100 LBF
MATERIALS	Ir-Re	BASE-LINE MATERIAL SYSTEM; OTHERS WILL BE INVESTIGATED IN TASK 4 AND BEYOND NO OTHER MATERIAL SYSTEM HAS DEMONSTRATED REQUIRED LIFE AT HIGH PERFORMANCE

Table 3.2-2

HIPC Parameter Space Bases for Parameter Selection

PARAMETER	SELECTION	BASIS
CHAMBER PRESS.	APPROX. 250 PSIA APPROX 500 PSIA	HIGHEST THAT CAN BE OBTAINED WITH EXISTING TANK PRESSURES HIGHEST THAT CAN BE PUMP-FED WITH POWER CONSTRAINTS TEST PROGRAM WILL COVER WIDER RANGE; WILL BE STRUCTURED SO THAT FLIGHT-TYPE ENGINE CAN BE EITHER HIGH OR MEDIUM Pc
MIXTURE RATIO	TBD	OPTIMUM FROM TESTING/ANALYSIS
VALVE	TEST STAND FLIGHT	TO PERMIT HIGH INLET PRESSURE OPERATION REQ'D FOR OPTION 3 FLIGHT TYPE ENGINE; MAY NEED FOR OPTION 2
INJECTOR	S/N LM-2 S/N 7 S/N 8	FOR TASK 2 TESTS USE EXISTING LASER-MACHINED (UNFIRED) INJECTOR DESIGN/FAB NEW INJECTOR FOR TASK 4 NEW DESIGN MAY BE REQUIRED AS ITERATION FOR OPTION 1 AND FOR FLIGHT-TYPE ENGINE BRILLIANT PEBBLES OR LDI EXISTING OPTIONS FOR 500 Pc ENGINE
FRONT END	FUEL-COOLED REGEN	APD HAS LARGE DATA BASE FOR 100LBF CLASS RUN INITIAL TESTS WITH WATER COOLING THERMAL BARRIER COATING WILL PROVIDE GOOD THERMAL MARGIN FOR N2H4
FLIGHT PRESSURIZATION SYSTEM	PRESSURE-FED PUMP-FED	FOR NEAR-TERM/REPLACEMENT AND GROWTH APPLICATIONS FOR LONG-TERM, HIGHEST PRESSURE APPLICATIONS ELECTRIC MOTOR DRIVE MONOPROP GG DRIVE
OTHER PARAMETERS	ENVELOPE, MASS, ETC.	TO BE FINALIZED AS PROGRAM PROGRESSES: USE COMPOSIT OF HUGHES/LMSC/GE/JPL/LORAL/ESA SPECIFICATIONS FOR INITIAL WORK

Table 3.2-2 (Cont.)

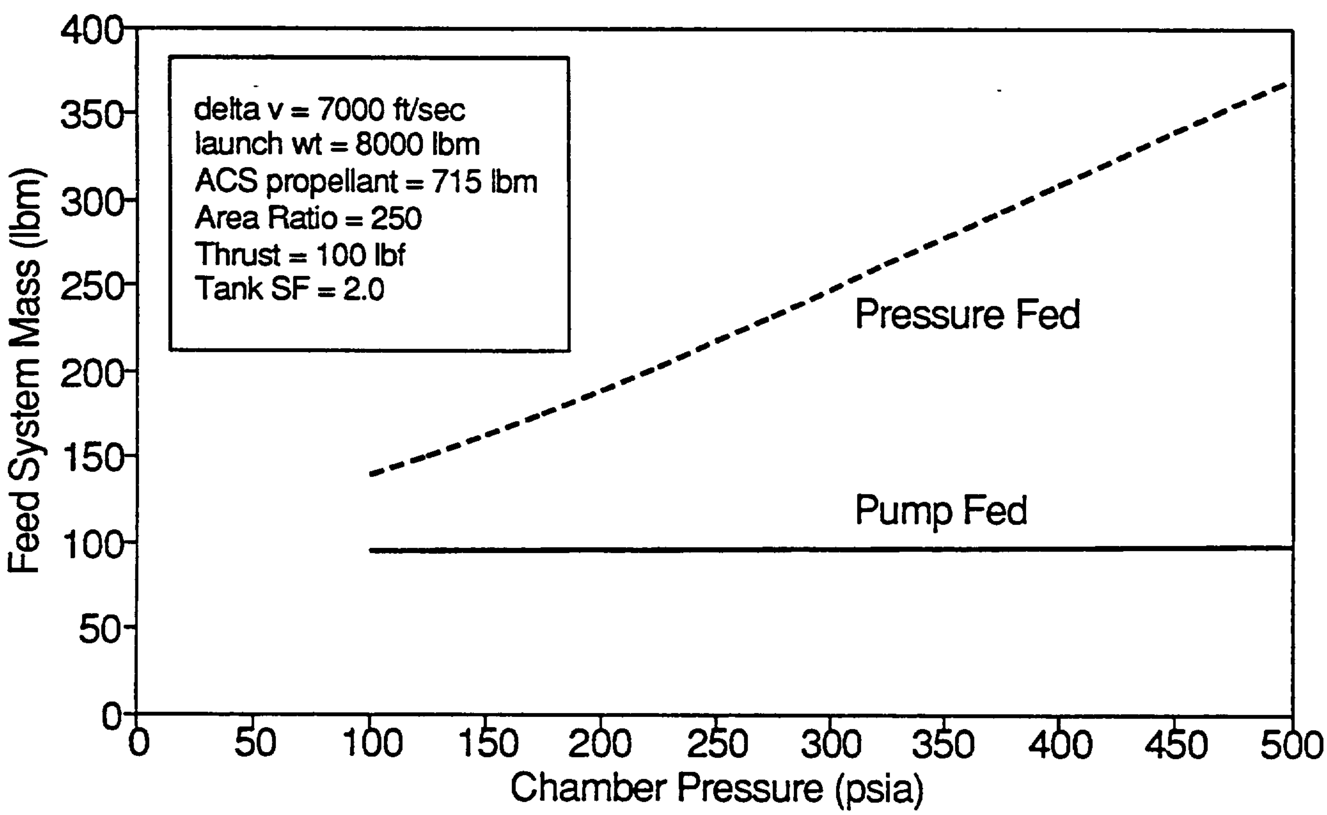


Figure 3.2.1-1. Large Feed System Mass is Associated With Pressure Fed Propulsion Systems

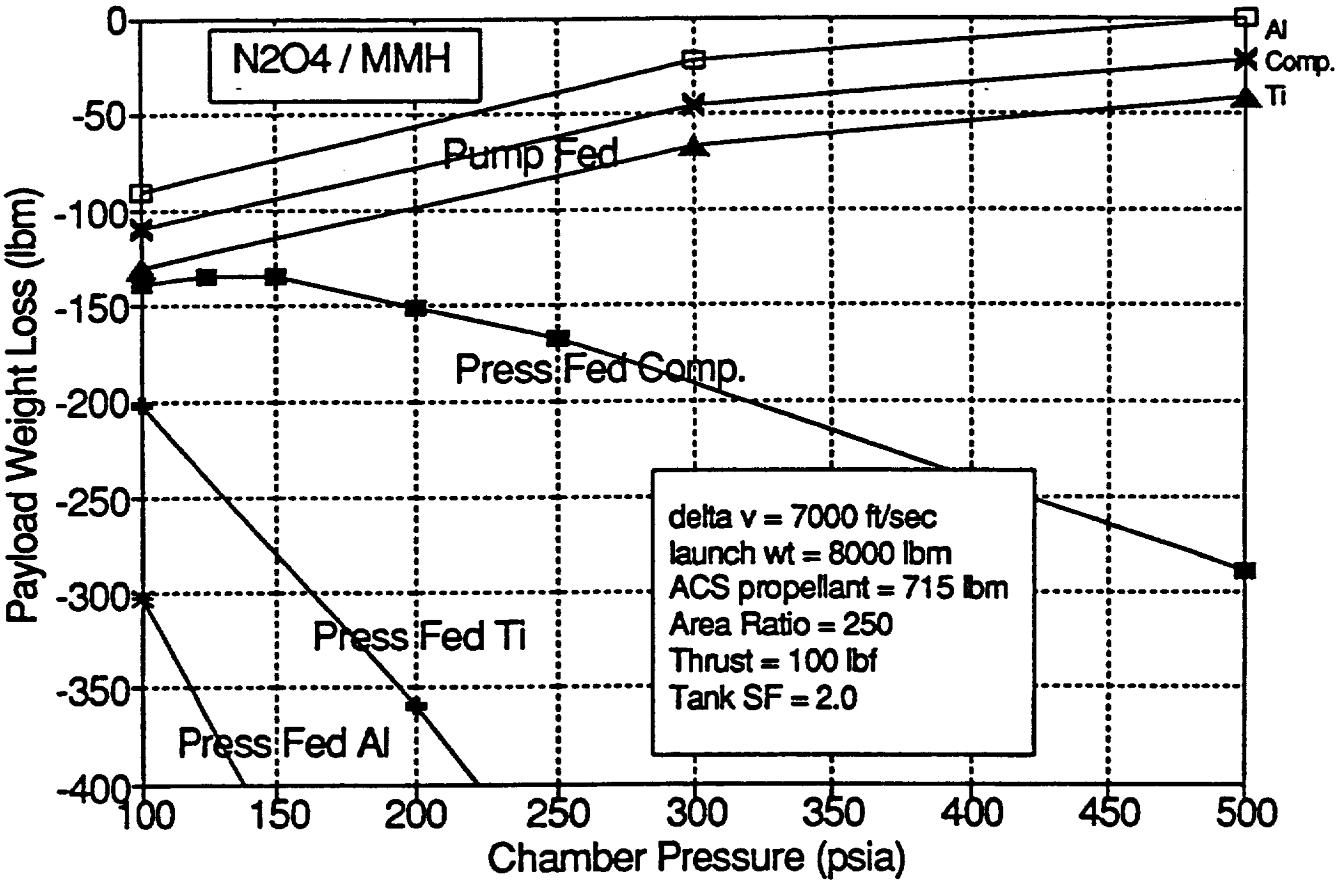


Figure 3.2.1-2. Intelsat 7 Payload Weight vs. Design

A range of design options, including pressurization system choice and tankage design were investigated to determine an optimum. These data are summarized in Figure 3.2.1-2, showing their effect on payload mass for an Intelsat 7-class craft using NTO/MMH; payload weight is shown relative to the best case: pump-fed, aluminum tanks, operating at 500 psia Pc. The study shows that the best pressure-fed system optimizes at about 135-150 psia Pc, with composite tanks, with about 140 lbm lower payload than the reference case.

The effect of changing from MMH to hydrazine for the same set of design conditions is shown in Figure 3.2.1-3. For this set of conditions there is a 17% increase in payload for the pump-fed system at 500 psia, and a 19% increase for the pressure-fed system. Also, the lower kinetics losses for hydrazine are evident in the pressure- fed system at low pressure, which optimizes at about 115 psia. The effect of chamber pressure on pump- and pressure-fed systems is illustrated in Figure 3.2.1.4 which compares spacecraft with gross low orbit weight (GLOW) of 1000 and 8000 lbm. As would be expected, the payload fractions are much larger for the larger system, by a factor of about 2.3

As a part of the ELES optimization, engine parameters are determined as a function of operating condition. The effect of chamber pressure on engine mass for a thruster designed to have minimum mass is shown in Figure 3.2.1-5 as a function of area ratio.

The effect of chamber pressure on engine envelope is shown as a function of area ratio in Figure 3.2.1-6. This in turn is used in the program to determine interstage, fairing and shielding weights.

It should be kept in mind that these studies are for 'flexible' spacecraft, which change design with assumed operating condition. For a predetermined spacecraft design, the optimum operating point for a pressure-fed system is at maximum design operating pressure for the existing tanks, which entails only an increase in helium supply system mass. On the other hand, for a pump-fed system, maximum Pc is set primarily by the amount of electrical power available for the pumps. The basis for chamber pressure selection is given in the next section.

3.2.2 Chamber Pressure Selection

A major objective of this program is to develop and demonstrate the technology for operation at high chamber pressure. "High" has not been defined, but is a resultant of what is technically feasible, what is 'salable' to spacecraft users, and is certainly higher than the 100+ psia range of conventional thrusters.

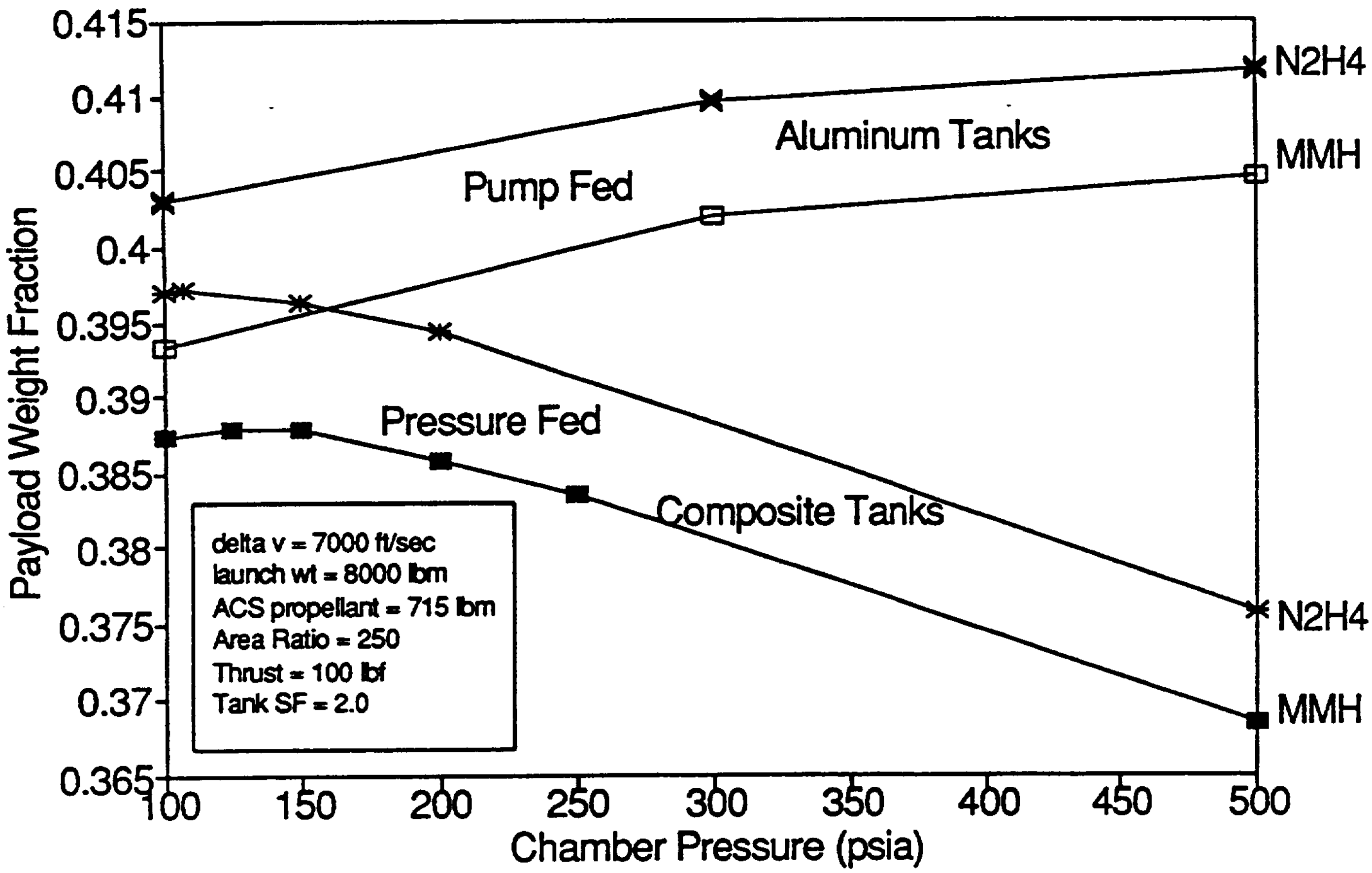


Figure 3.2.1-3. Intelsat 7 Payload Weight vs. Design

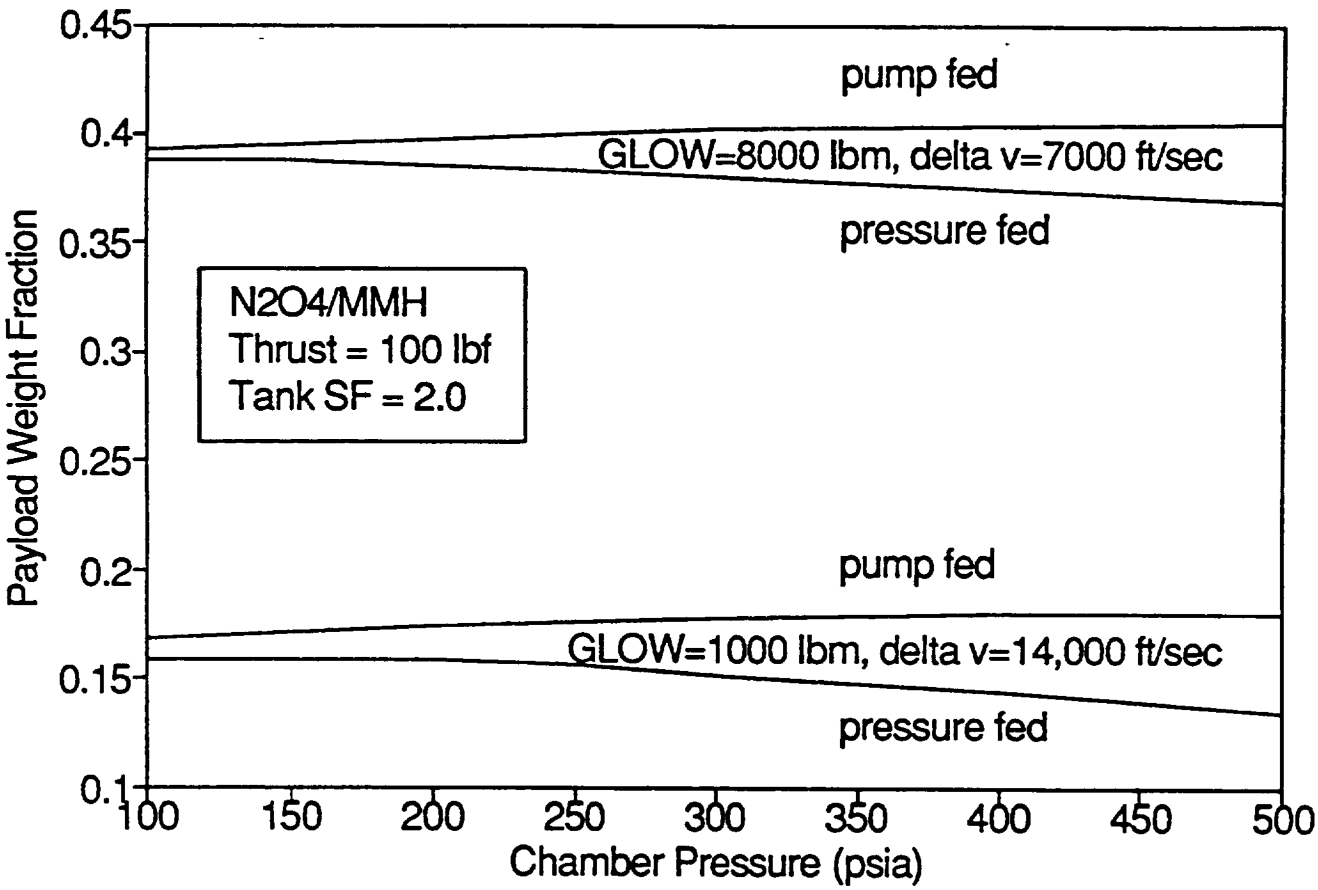


Figure 3.2.1-4. Comparison of Small and Large Systems

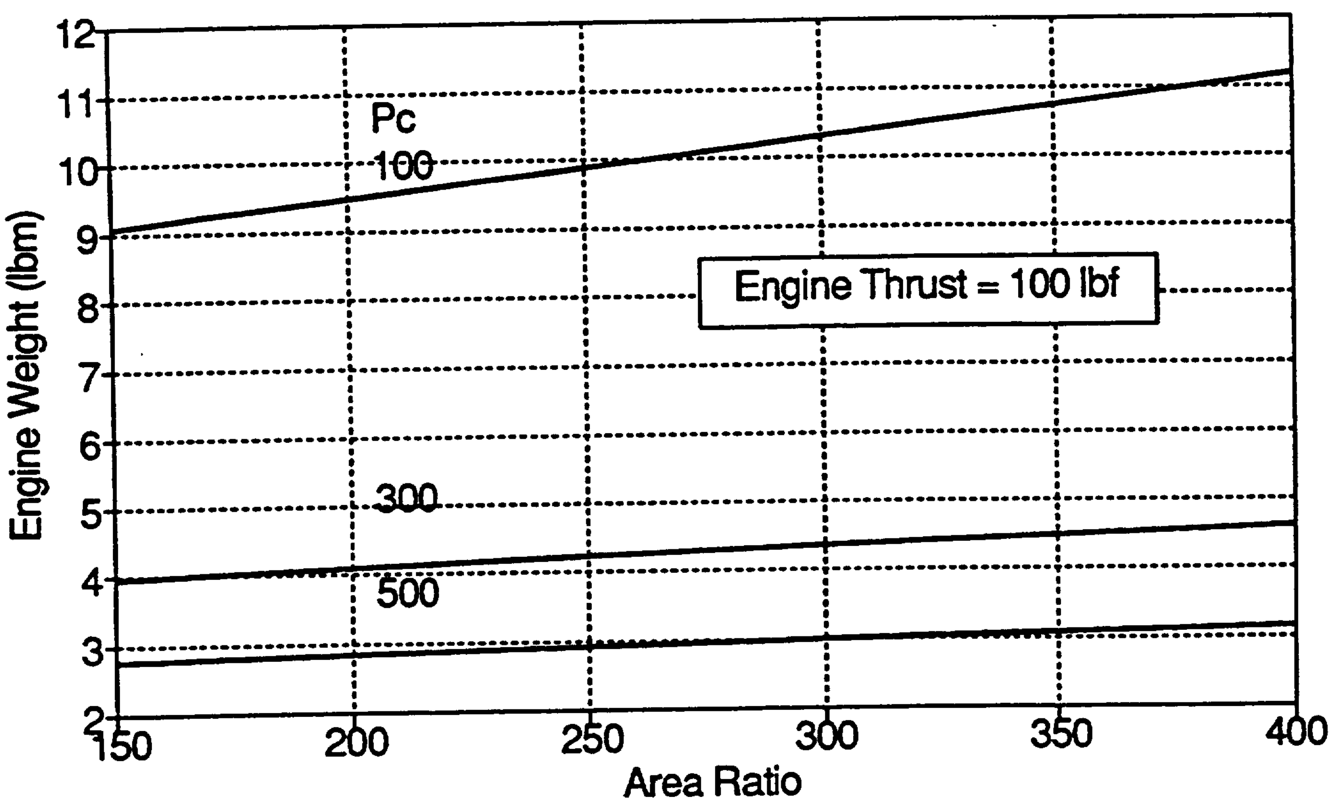


Figure 3.2.1-5. High Pc Gives Low Engine Weight (Includes Valve, Injector, Chamber, Nozzle)

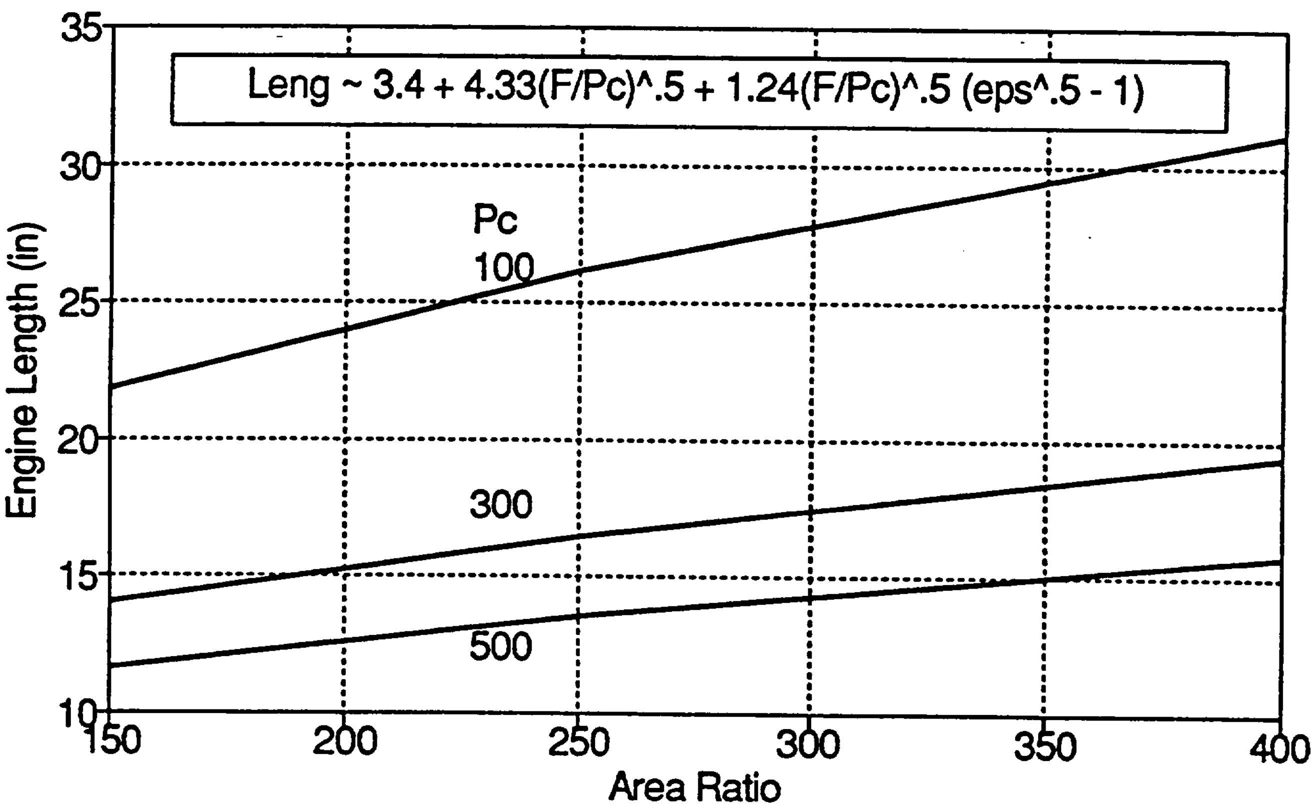


Figure 3.2.1-6. Engine Length is Inverse With Pc (for Thrust = 100 lbf)

As shown is the previous section, with a conventional pressure-fed system, optimum chamber pressure is at about 150 psia for a reference Intelsat 7-type spacecraft. With pump-fed systems payload is still increasing slightly with pressure at 500 psia. The pressure-fed analysis assumes tanks are designed for the specified operating chamber pressure, while the pump-fed case assumes minimum gage tanks operating at the required suction inlet pressure.

In fact, pressure-fed spacecraft propulsion systems generally do not take advantage of the tank capabilities. Although capable of safe working pressures of typically 400 psia, they are routinely operated at lower pressures and often over a very wide pressure range; a typical example of this is the Bus-1 tank pressure schedule (Fig. 3.2.2-1), from Ref. 6. This eases the job of the spacecraft designer, but makes it difficult for the propulsion designer to deliver maximum performance. This pressure range is the result of 1) use of a blow-down system with few or no repressurizations, 2) use of on-off pressurization valves (to avoid unreliable regulators) with a wide reset band, or 3) use of regulators with a wide dead band.

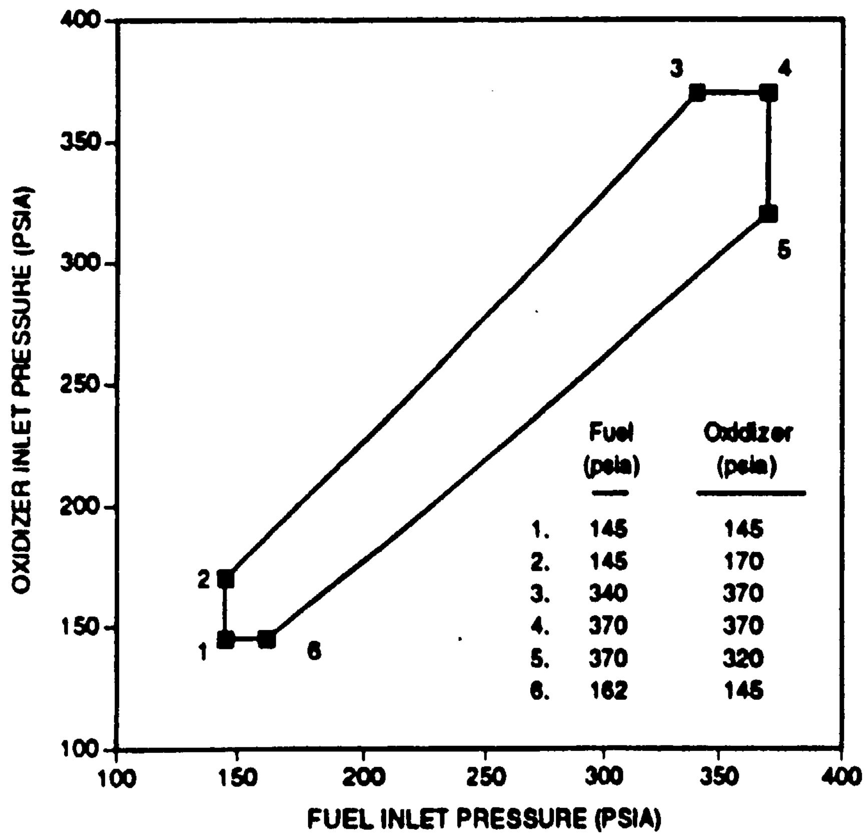
There are two limiting applications for high pressure thrusters: those which use existing pressure-fed propellant delivery systems, and those which would use low pressure tankage and pump-fed systems.

<u>Pressure-Fed</u>

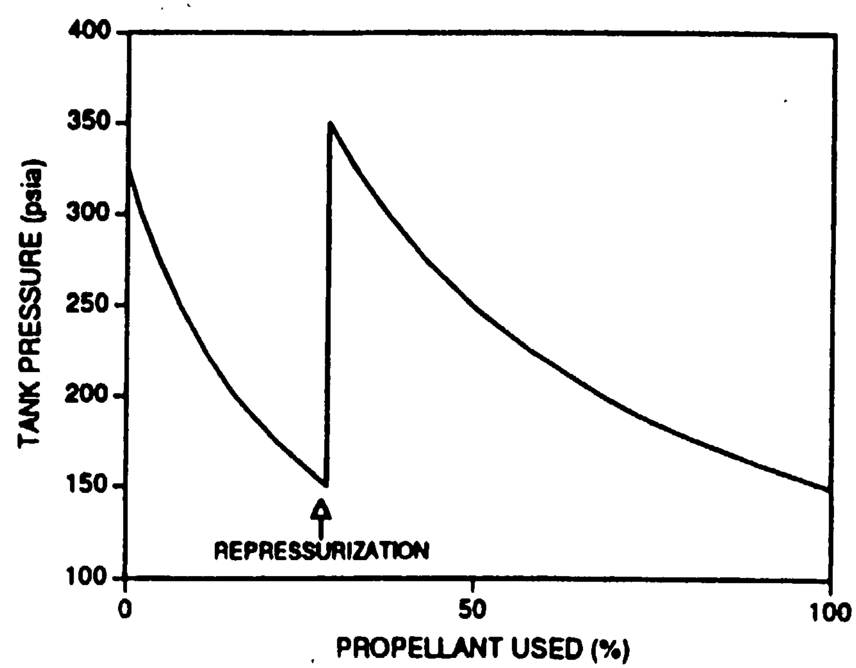
Existing propellant tankage can supply propellants to thrusters operating at significantly higher chamber pressure with relatively minor changes to the pressurization system and no structural changes. Table 3.2.2-1 compares operating pressures of several spacecraft propellant delivery systems and the corresponding maximum chamber pressure potential. The thruster pressure schedule which would permit this shows that the upper practical limit for this approach is about 250 psia chamber pressure. To operate in this mode would require tight control of tank pressure (+-20 psi is practical). Since the thruster would necessitate a low delta P injector design, it would be closer to its chug instability limit, as discussed in Section 3.3.3.

<u>Pump-Fed</u>

Numerous pumping schemes have been considered for this application. Pump power in the range of several horsepower is required. Gas generators, topping cycles or preburner cycles involve too much performance loss and added complexity at this thrust level. The only other potential pump power source on board the spacecraft is electrical, from the solar panels. As seen in the survey results (Section 3.1.2) electrical power in the 2 to 3 kw range is



Thruster Operating Envelope



Pressure Blowdown Profile

Figure 3.2.2-1. BUS-1 Propulsion System Pressure Schedule

Table 3.2.2-1

System Selection Parameter – Chamber Pressure

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SYSTEM SELECTION PARAMETER-- CHAMBER PRESSURE

SYSTEM	NOMINAL INLET PRESSURE, PSIA	NOMINAL TANK PRESSURE, PSIA	NOMINAL CHAMBER PRESSURE, PSIA	MAX TANK PRESSURE, PSIA	NOMINAL DESIGN APPROX MAX CHAMBER PRESSURE PSIA	LOW DELTA DESIGN APPROX MAX CHAMBER PRESSURE PSIA
JPL/CASSINI	235	250	115	330	185	220
LMSC	235	250	115	370	225	260
HYPOTHETICAL	235	250	115	400	255	290

APPROACH:

o MODIFY PRESSURIZATION SYSTEM TO USE REGULATOR OR BANG-BANG VALVE TO MAINTAIN TANK PRESSURE +-10PSI

o MODIFY INJECTOR AND PLUMBING TO REDUCE THRUSTER PRESSURE DROP

o NOMINALLY PERMITS OPERATION WITH MINIMUM CHANGE IN PROPELLANT SUPPLY SYSTEM; MAY REQUIRE ADDED HELIUM.

practical now on many systems. In the future, arrangements could be made to provide more. The pumping power requirements for a specific engine system are shown in Figure 3.2.2-2, which shows that a chamber pressure of 500 psia could be achieved with typical available power levels.

In summary, we have defined two levels of high chamber pressure: 250 psia for thrusters to be fitted to existing spacecraft propellant delivery systems, and 500 psig for new spacecraft where tankage and propellant feed system design is open.

3.2.3 Propellant Selection

Given the need to employ earth-storable propellants with high performance, a relatively restricted set of choices is available. These possible combinations, along with two common and one exotic cryogenic combinations for comparison, are shown in Table 3.2.3-1.

The table compares one-dimensional equilibrium (ODE) performance at common reference conditions. Actual delivered performance rankings change significantly, as will be discussed more fully in Section 3.3.1, Performance Determination.

All of the storable propellants are high in cost relative to cryogenics, so there is no clear cost discriminator here for selection. The survey responses for NTO/AH or MMH and ClF5/AH are shown (neither H_2O_2/AH nor the cryogenics were given as options in our survey). The survey responses were nearly an even split on hydrazine or MMH as fuels; no one selected ClF5 as their preferred oxidizer.

User acceptance is good for either hydrazine or MMH with NTO; it is nil for the other propellants in the applications of interest.

Some logistics impacts of propellant choice which are specific to ease of conducting our hot fire testing are shown in the table. Costs of testing are minimized if they can be conducted in our A-Area small thruster facility; this is practical for both N_2H_4 or MMH. ClF_5 testing must be conducted in the J-Area; environmental regulations limit us to a maximum of 22 lbm ClF_5/day. This corresponds to about 110 sec of firing per day with ClF_5 oxidizer, while the storage limits in the A-Area permit 1600 sec/day with NTO oxidizer. Ultimately the infrastructure, i.e. environmental permits, propellant handling and production capability may be developed to permit conduct of a program such as this with ClF_5; at present they are not.

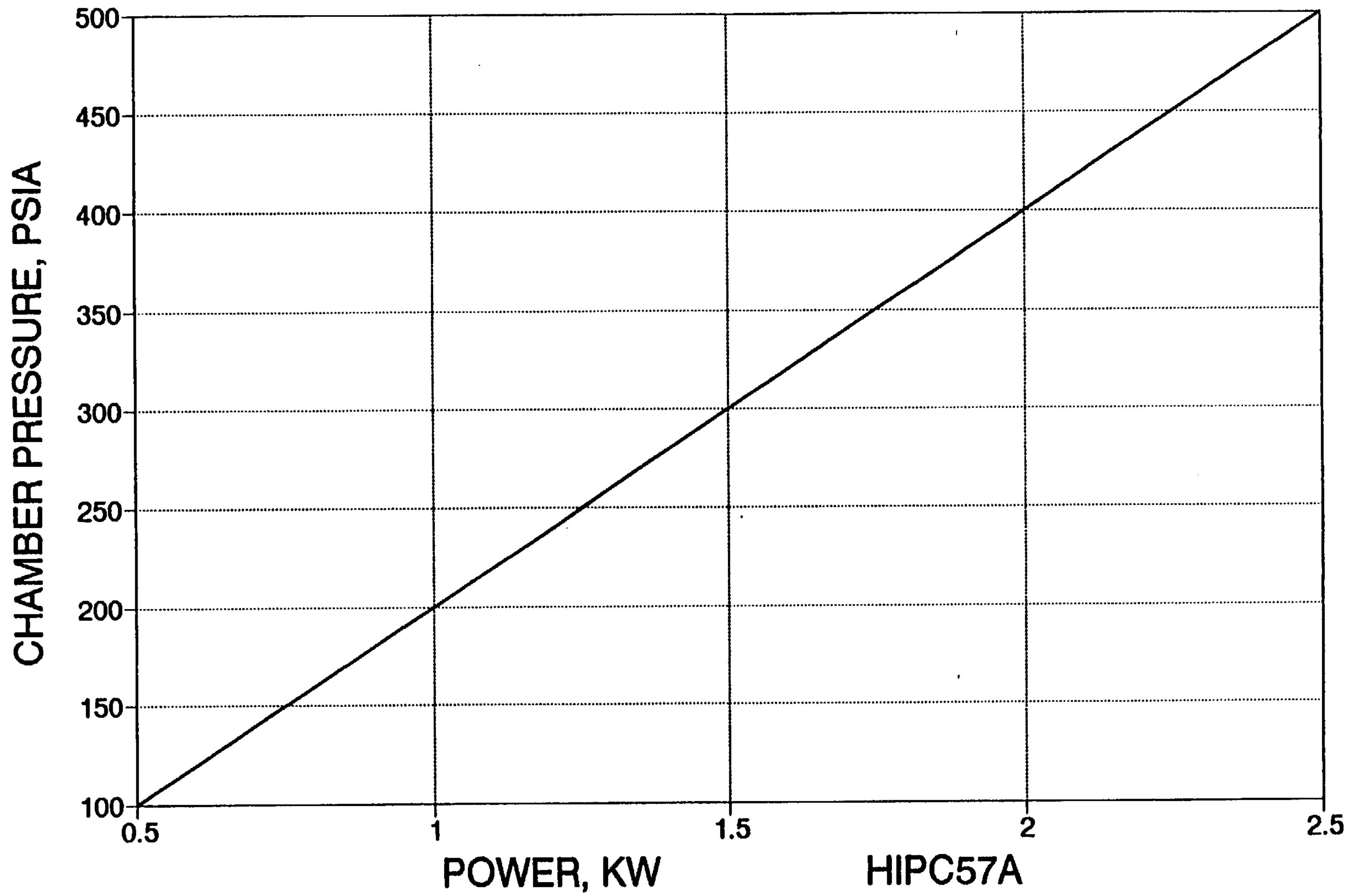

Figure 3.2.2-2. Allowable Chamber Press vs. Power Available, F= 100 lbf

Table 3.2.3-1

HIPC Parameter Space Potential Options
for Propellant Choices

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HIPC PARAMETER SPACE

POTENTIAL OPTIONS FOR PROPELLANT CHOICES

PROPELLANT OPTIONS	SPECIFIC IMPULSE [1]	AT MR=	EARTH STORABLE	COST	SURVEY NUMBER PREFERRING	USER ACCEPTANCE	TEST AREA OF CHOICE	LIMITING PROP. AMOUNT	TEST FACILITY DURATION LIMITS SEC/DAY[2]	MEETS RFP REQUIRE-MENTS?	COMMENT
N2O4/N2H4	343.8	1.42	YES	HIGH	13	GOOD	A-AREA	30 GAL	1600	YES	DELIVERED Is>330 SEC
N2O4/MMH	341.5	2.37	YES	HIGH	16	GOOD	A-AREA	30 GAL	1600	YES	HOWEVER, DELIVERED Is<330SEC
ClF5/N2H4	365.3	2.71	YES	HIGH	0	NONE	J-AREA	22 LBM OX	110	NO	NO USER ACCEPTANCE, MINIMUM FACILITY INFRA-STRUCTURE, FACTOR OF 10 INCREASE IN TEST COSTS
N2F4/N2H4	390.5	3.25	NO	HIGH	[3]	NONE	J-AREA	TBD	TBS	NO	NOT EARTH STORABLE
H2O2/N2H4	337.6	2.12	YES	HIGH	[3]	NONE	A-AREA (?)	30 GAL?	TBS	NO	DELIVERED Is<330SEC
O2/H2	455.3	4.83	NO	LOW	[3]	NONE	A-AREA			NO	NOT EARTH STORABLE
O2/CH4	368.9	3.45	NO	LOW	[3]	NONE	A-AREA			NO	NOT EARTH STORABLE

[1] ODE, Pc=1000, e=40:1
[2] BASED ON EXISTING STORAGE/EMISSION PERMITS
[3] NOT AN SURVEY OPTION

As indicated in the table NTO with either MMH or AH meet the program basic requirements. However, considering the time frame of this program, the nitrogen tetroxide-hydrazine combination has been chosen because of its potential of providing an Is greater than 330 sec while allowing both dual mode operation and electrical augmentation.

Choice of hydrazine has a moderate impact on propellant costs relative to MMH, both because of the slightly higher unit cost and the larger amount required at its optimum mixture ratio. Programmatic effects of propellant change are discussed in Section 4.

3.2.4 Thrust Class Selection

As shown in the survey results the majority of the users expressed a preference for the 100 lbf class thruster for their spacecraft. In addition, recent procurement activity has emphasized this thrust level (Table 3.1.3-1). Finally, as will be shown, the preferred applications require large total impulse which convert to unrealistically long burn times at lower thrust levels.

Choice of 100 lbf as the design point rather than the 14 lbf thruster (which would have been operated at a nominal thrust of 22.5 lbf for this program) has an impact on propellant usage that will be minimized by reducing testing and hardware fabrication where practical, as discussed in Section 4.

3.2.5 Total Impulse Selection

Total impulse, thrust level, total quantity of propellant burned, and maximum reliable total burn time are interrelated as shown in Figure 3.2.5-1. This figure also shows propellant quantities for three spacecraft: Iridium, the Hughes-601 bus, and the LMSC Bus-1.

The figure also shows the presently demonstrated maximum burn time of 15 hours for the Ir-Re system. At 100 lbf the Hughes- and LMSC-class engines will require less than 15 hours in flight; it should be noted that the spacecraft manufacturers typically require demonstration in Qualification testing of 150% of maximum expected flight burn time. Excessively long burn times would be required for the 15 lbf-class thruster.

3.2.6 Engine Cost Considerations

Recent experience with competitive procurements for this class of engine show that cost is given more emphasis than would be indicated by the user survey. The

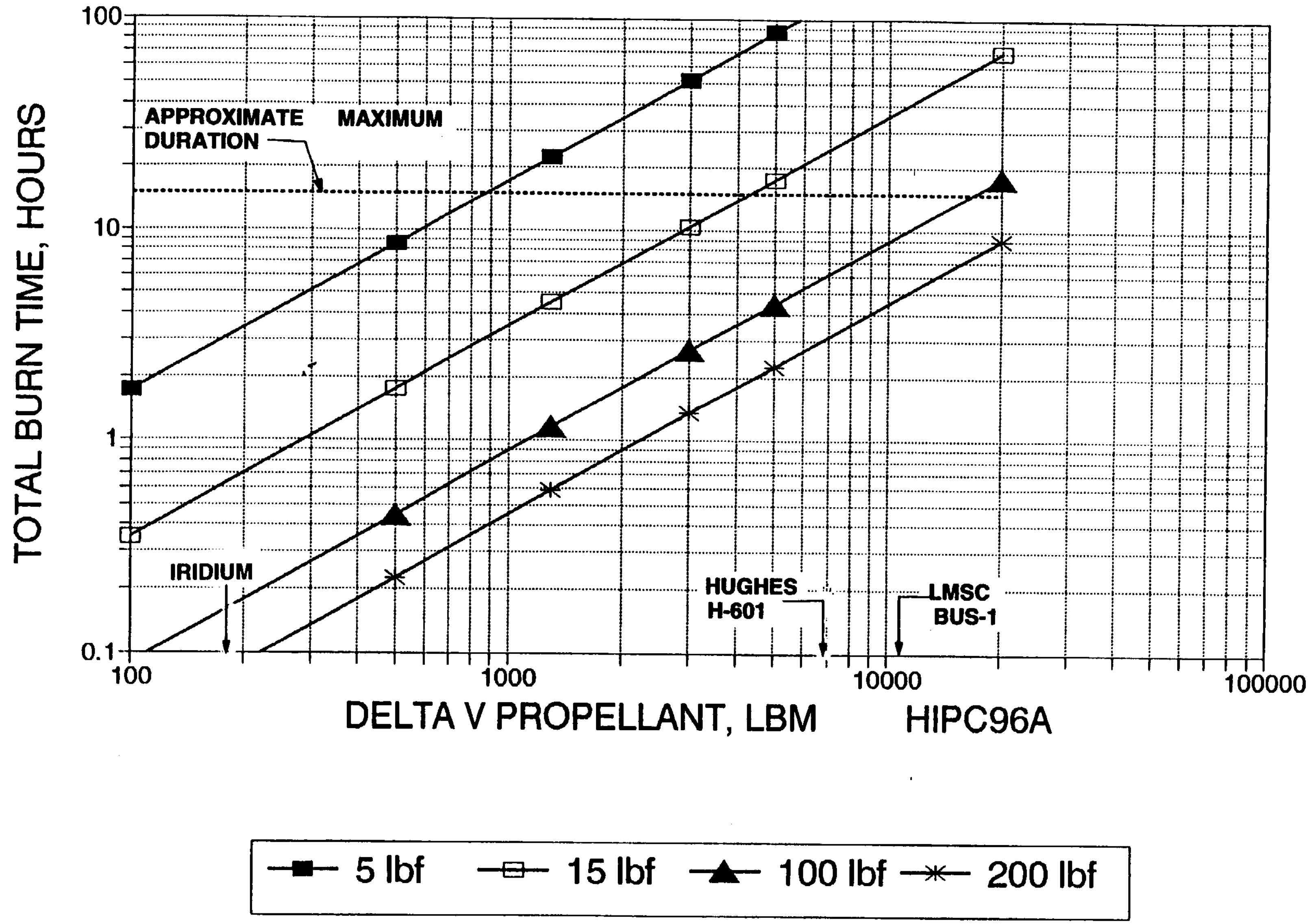

Figure 3.2.5-1. Required Total Burn Times

conventional thruster of this type uses silicide-coated Cb. Because of significant differences in cost of the basic raw materials, Ir-Re chambers must always be more expensive than Cb chambers. This is illustrated in Figure 3.2.6-1, which plots basic raw material costs for seven different chamber material systems. These are small quantity prices and do not reflect the cost of the actual precursor required or the fabrication costs.

During cost studies made on a recent Aerojet program (Ref. 7), the cost of the reference AJ10-221 Ir-Re engine was analyzed, both with "conventional" and low cost approaches. The relative cost of the six factors which make up the engine expense: chamber, valve, injector, nozzle, component assembly, and acceptance test are shown in Figure 3.2.6-2. This shows that by far the largest cost factor is the chamber, about 45% of the total engine cost. Since in principle the valve, nozzle and test could be the same as for a conventional engine, the chamber is obviously the area which must be worked the hardest and will get no benefits from further cost reductions in conventional engines.

Cost Savings

The unavoidable higher costs of the Ir-Re thruster are offset by the life-cycle cost savings which result from its higher performance. In the spacecraft applications, savings can take one or a combination of three forms. The higher performance can provide more spacecraft life due to the availability of more station-keeping/attitude control propellant at orbit insertion. The effect of this first form of revenue enhancement is shown in Figure 3.2.6-3. Increased revenue relative to normal spacecraft life (based on a Cb chamber delivering Is=315 sec) is shown for four engine configurations. The "standard" AJ10-221 Ir-Re chamber, operating with NTO/MMH at Pc=100 psia would increase total revenue from transponder leases by 10%. Changing to hydrazine with the same chamber pressure would result in a revenue increase over the baseline of about 18%. Operating this propellant at a chamber pressure of 250 psia would give a 22% increase, while a 30% increase would result for operation at 500 psia.

Because no savings are realized until the end of the normal spacecraft life, because of the uncertainties in projecting the likelihood of shortened life due to other subsystem problems, and because of the unknown demand for transponder channels ten years hence, this approach to revenue enhancement is not attractive to company financial officers.

The second revenue enhancement approach is to plan on normal design life for the spacecraft and to obtain a rebate for the off-loaded excess propellant. For Ariane this savings is about $10,000/lbm. The magnitude of this savings, which is much less than that

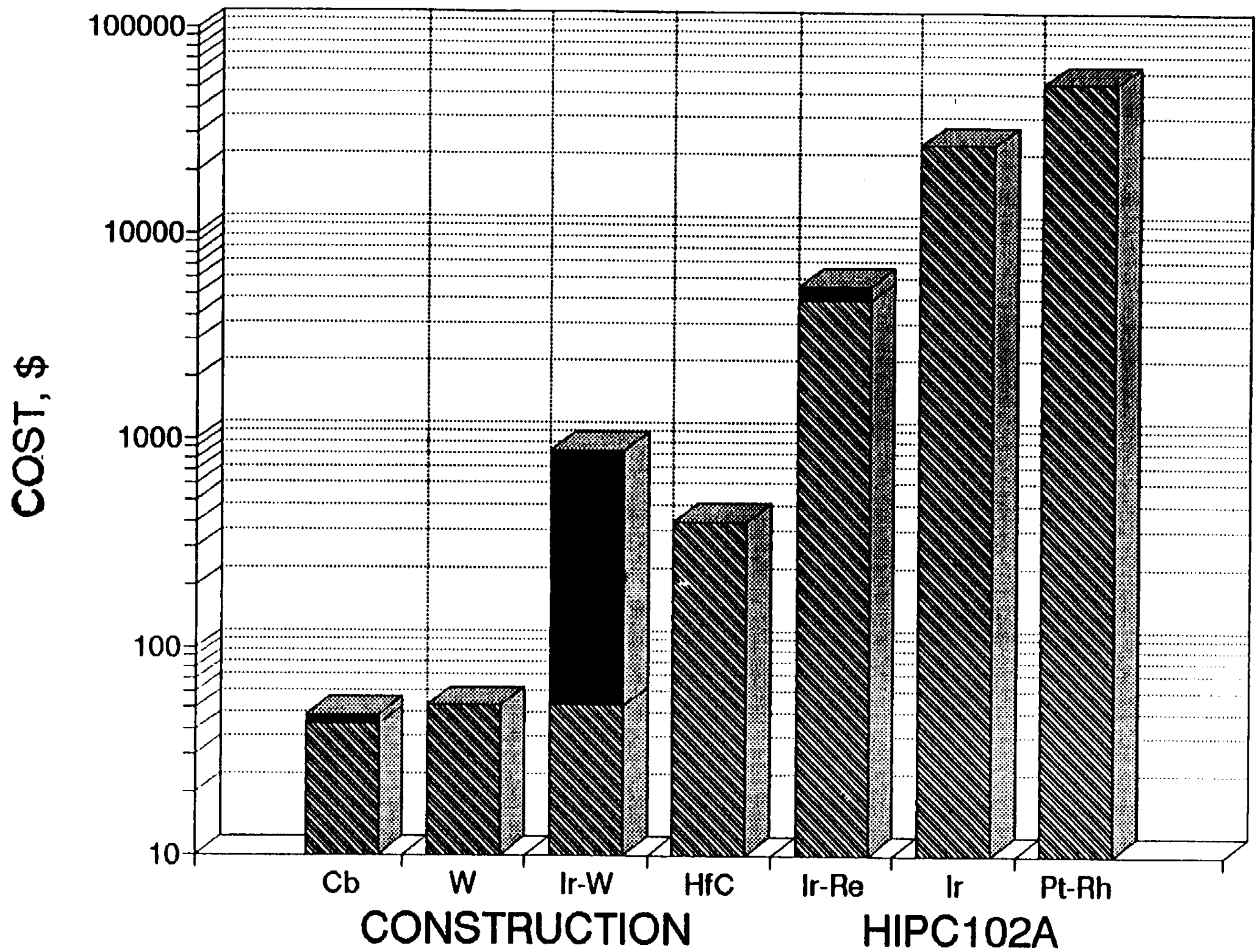

Figure 3.2.6-1. Chamber Basic Raw Material Costs, Small Quantity Prices

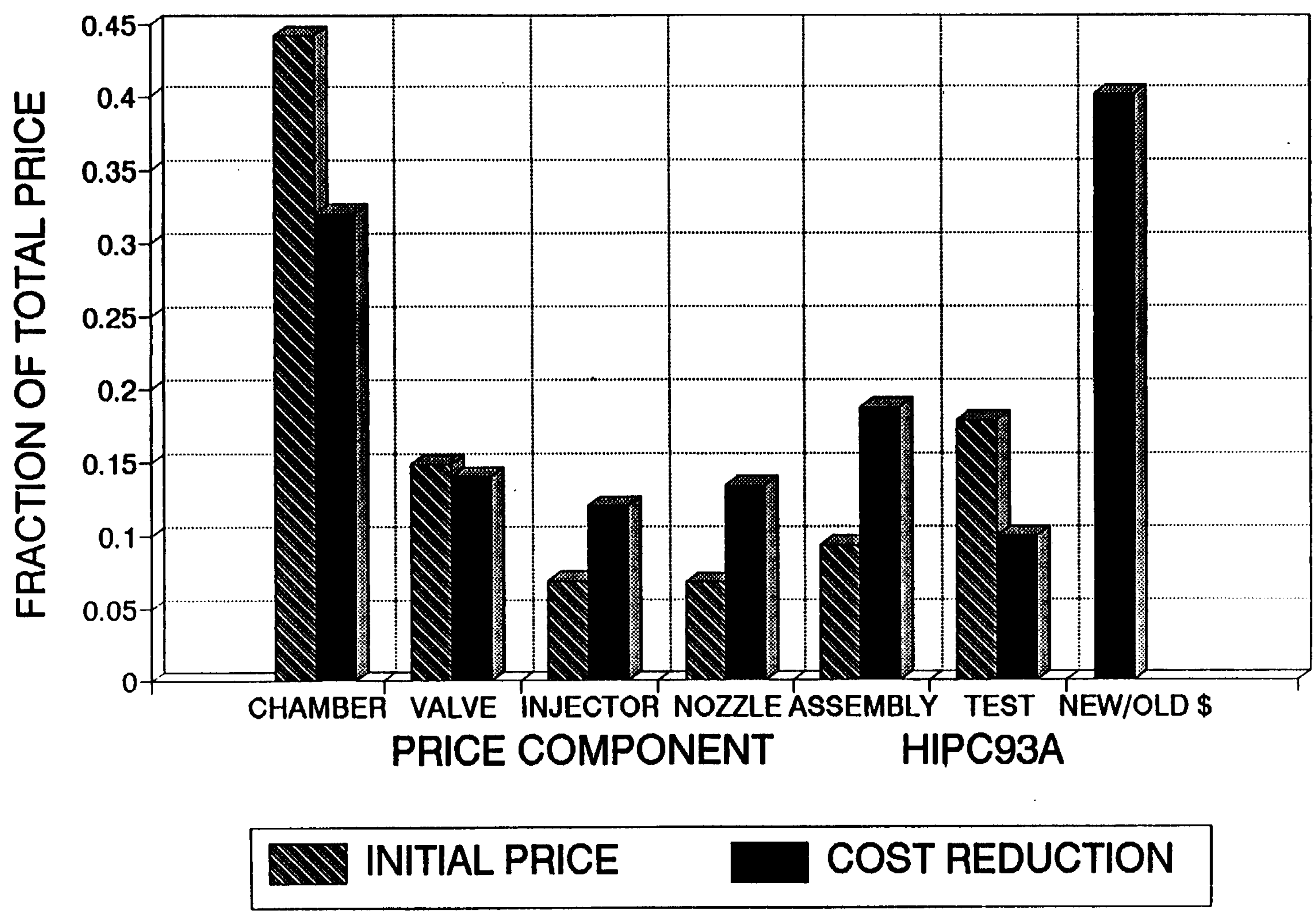

Figure 3.2.6-2. Normalized Ir-Re Engine Price, Improved AJ10-221 Design

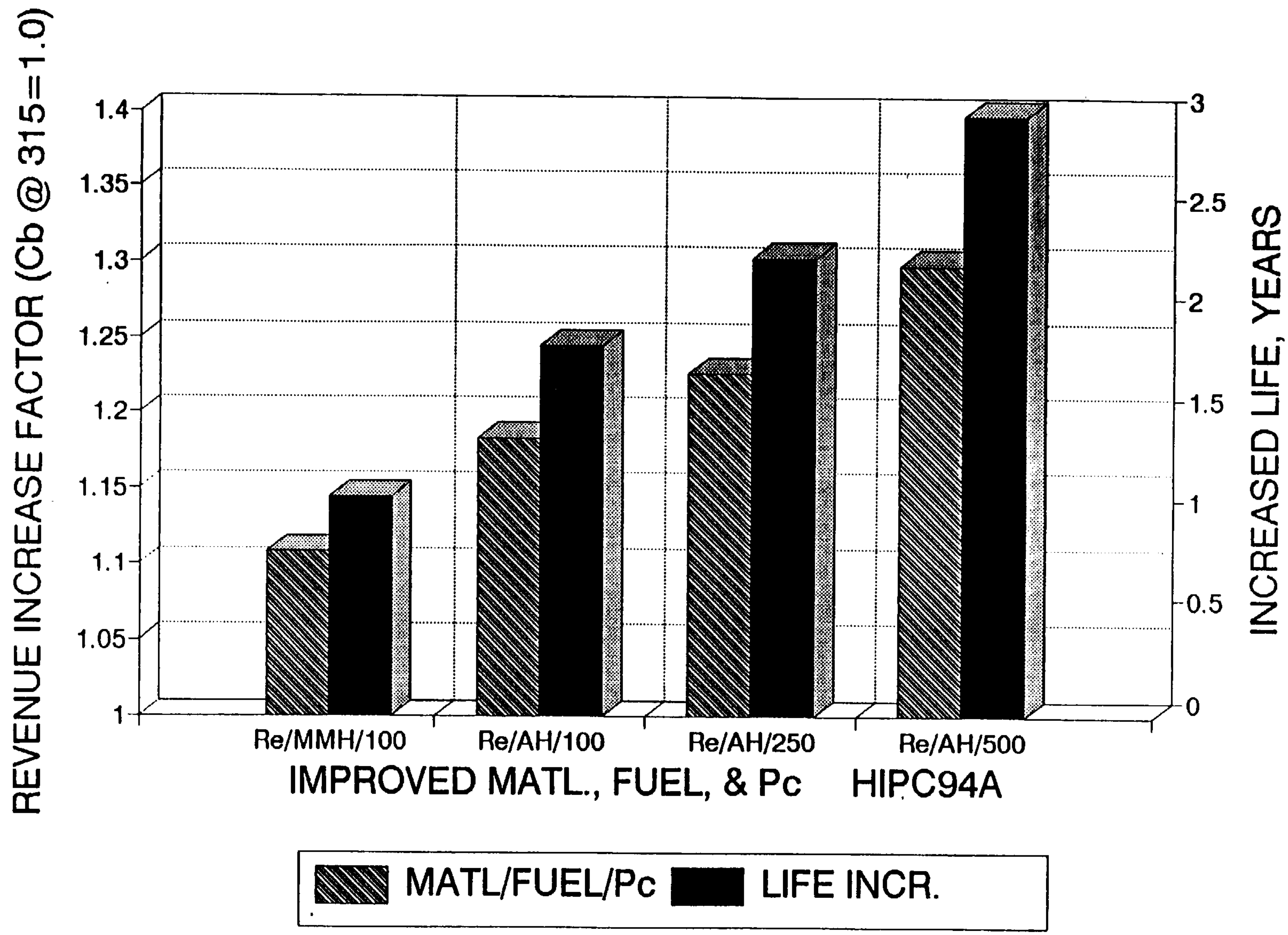

Figure 3.2.6-3. Revenue Increase By Life Extension, Comm. Sat.; Base = 10 yrs, Cb Chamber

achievable from longer life, is shown in Figure 3.2.6-4, as a function of total impulse of the spacecraft. The same four advanced thruster cases are compared to a conventional Cb engine at Is=315. Three typical spacecraft, Iridium, H-601, and Bus-1 are indicated on the graph. The trivial savings for the Iridium case indicates why that class of spacecraft is not a suitable candidate for the high performance thruster, from the standpoint of cost savings.

The incentive to the launch agency for making this rebate is that it may make it possible to put an additional (paying) spacecraft on board. Since this is an upfront return to the spacecraft manufacturer, obtained regardless of outcome of the launch or life of the craft, this option is expected to be selected.

The third class of savings relates to cases where a prime can off-load sufficient mass so that launch on a less expensive vehicle is possible, such as dropping from a Titan to an improved Atlas. This is more difficult to quantify but is a definite option.

3.3 CONCEPTUAL DESIGN CRITERIA

The application/parameter space choices discussed in Sections 3.1 and 3.2 have been used to develop design concepts for high-pressure, earth-storable flight thrusters. Since two ranges of potential Pc have been identified as practical, two concepts have been prepared. These concepts are compared to our reference engine, the AJ10-221 Ir-Re, 286:1, 490 N engine developed in Ref 8.

The purpose of these concepts is to provide a signpost to guide the program towards Task 11, Option 3, where the flight thrusters will be designed, starting in December of 1996.

The appearance of the two concepts relative to the AJ10-221 is shown to the same scale in Figure 3.3-1. The details of the thruster designs, their operating conditions, and expected performance are shown in Table 3.3-1. The two high pressure concepts have been labeled 1A (250 psia Pc), and 2A (500 psia Pc). The chamber pressure increase has resulted in an over-all decrease in thruster size. As will be discussed in the stability section, the 250 Pc case has not shrunk in chamber dimension because stability considerations require that sufficient chamber volume be provided to maintain stability at its relatively low injector pressure drop. Conceptually, the 500 psia case has been allowed more pressure drop, although it is constrained by available electric power.

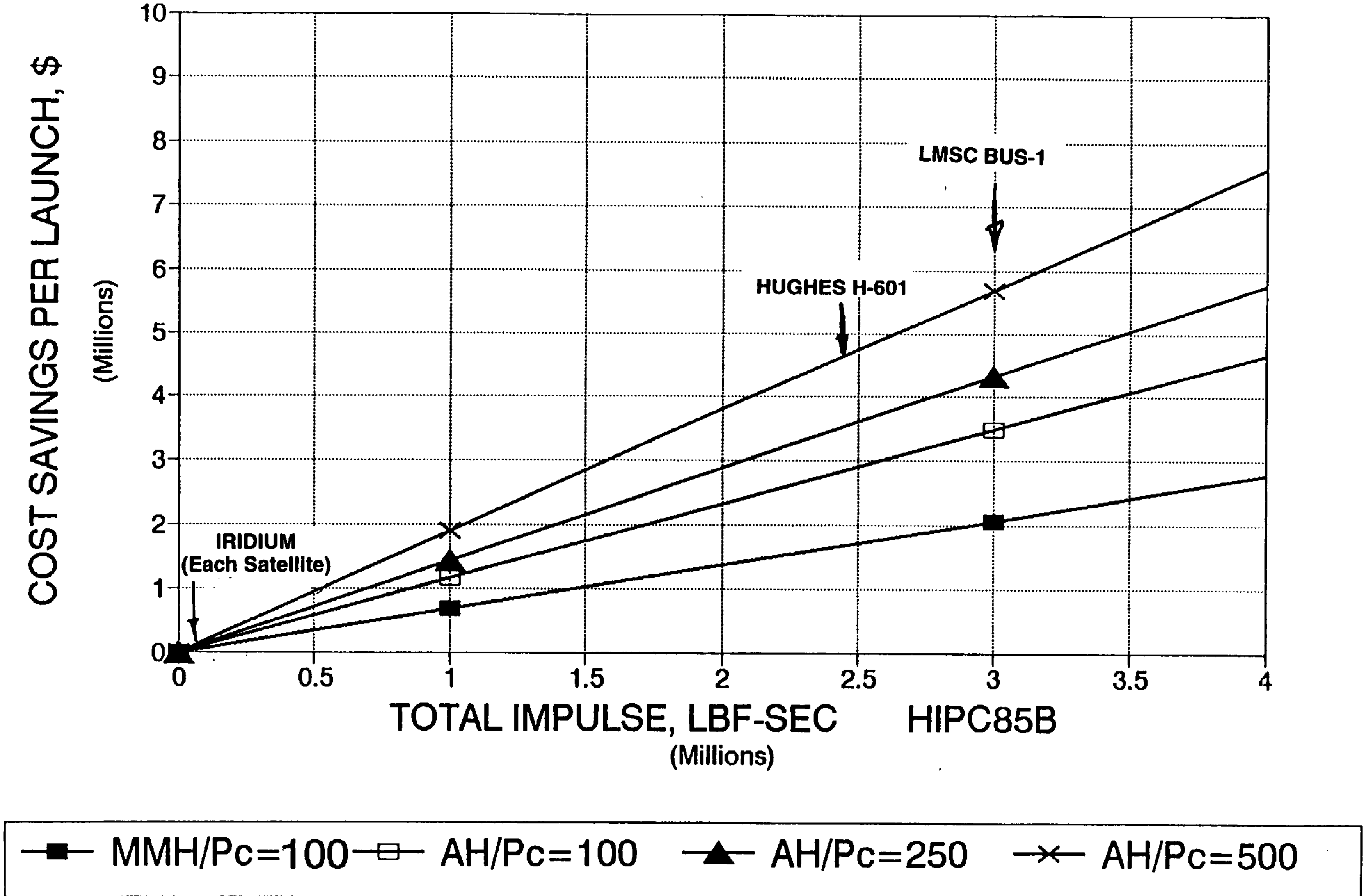

Figure 3.2.6-4. Cost Savings Ir-Re vs Conventional, Base - MMH-Cb, 100 Pc; $10,000/lb Off Load

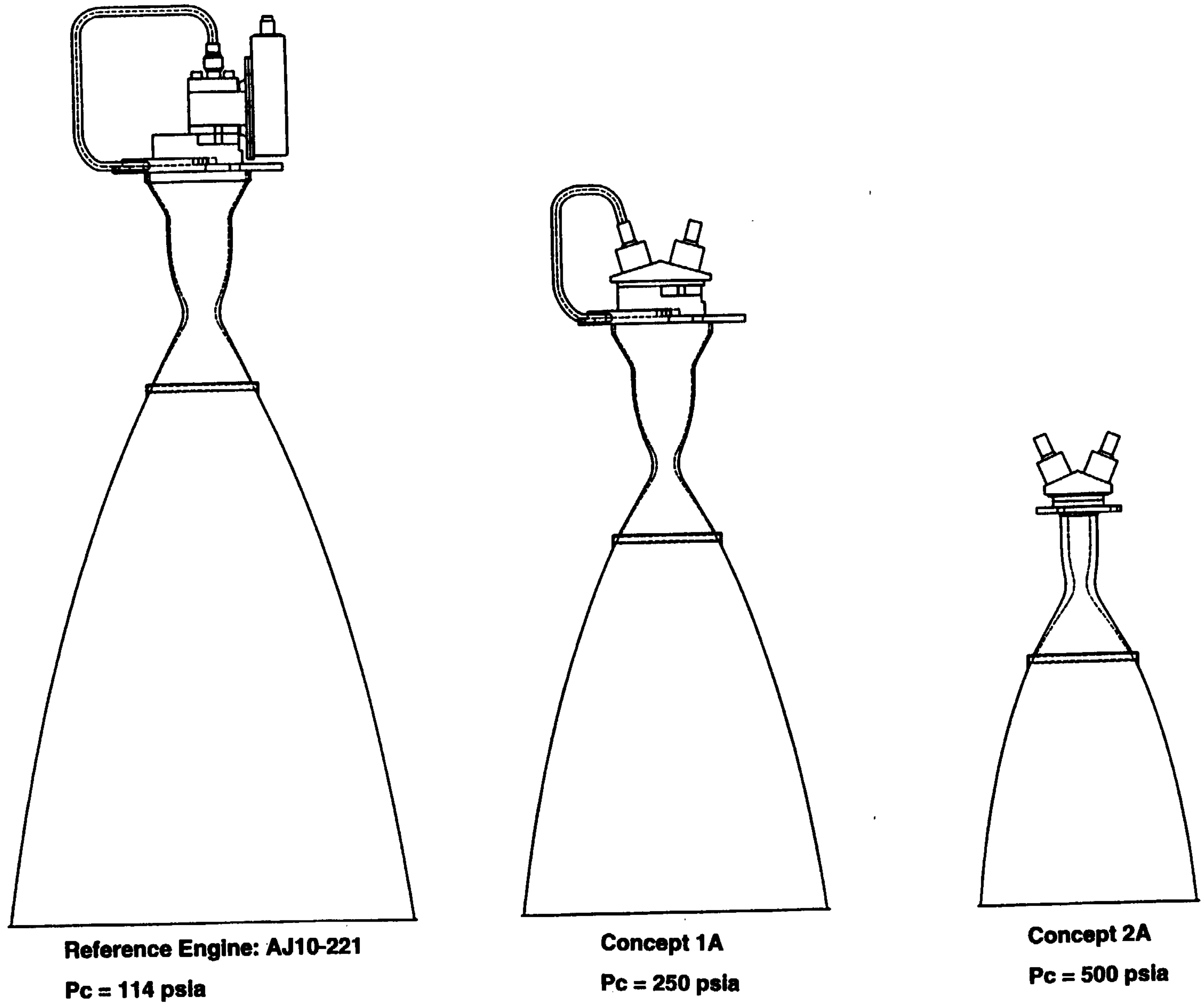

Figure 3.3-1. Reference and Flight Concept Rocket Engines

BASIS FOR CONCEPTUAL DESIGNS

FLIGHT TYPE CONCEPT DESIGN=	AJ10-221 [REFERENCE]	#1A	#2A
CHAMBER PRESSURE SELECTION CRITERIA	JPL SPEC	~MAX. Pc WITH EXISTING TANKS[1]	~MAX. Pc PUMP-FED[2]
DESIGN Pc, PSIA	115	250	500
THRUST, LBF	110	100	100
PROPELLANTS	NTO/MMH	NTO/HYDRAZINE	NTO/HYDRAZINE
DESIGN BASIS	JPL SPEC	AJ10-221	BRILLANT PEBBLES
Is, SEC	321.8	330	335
C*, FT/SEC	5500	5650	5720
NORMALIZED Is PERFORMANCE [AJ10-221=1.0]	1	1.025	1.041
THRUSTER EFFICIENCY [3]	0.91	.93	.94
NORMALIZED CHAMBER HEAT FLUX [AJ10-221=1.0]	1	0.93	6.8
NORMALIZED THROAT HEAT FLUX [AJ10-221=1.0]	1	1.55	2.36
CHAMBER TEMPERATURE, oF	3380	3790	3950
ENVELOPE, MAX DIA., IN	13.8	9.2	6.5
ENVELOPE, MAX LENGTH	30	20.7	15
MAX WEIGHT, LBM	10	TBD	TBD
VALVE	MOOG TORQUEMOTOR	LOW COST	LOW COST
AREA RATIO	286	300	300
DESIGN LIFE, HOURS	>6	>12	>12
FRONT END DESIGN	FUEL-REGEN S.S.	FUEL-REGEN S.S.+THER. BARR. COATING	FILM-COOLED PLATINUM-Rh TRIP
INJECTOR	S/N6-2	RE-BALANCE AJ10-221 FOR HYDRAZINE	USE BRILLIANT PEBBLES TI INJ.
CONTRACTION RATIO	4	11	2.4
L', IN	4.2	4.2	2.1
THROAT DIA, IN	0.804	0.521	0.368
CHAMBER DIA, IN	1.71	1.71	0.57
MINIMUM INJECTOR DELTA P, PSI [STAB. LIMIT]	35 (CALC)	75	150
CHAMBER MATERIAL	Ir-Re	LOW-COST Ir-Re	LOW-COST Ir-Re

[1] TYPICAL MAX. PRESSURE FOR EXISTING TANKS IS ~400 PSIA.
[2] POWER LIMITED AT ABOUT 3 KW.
[3] RELATIVE TO ODE AT DESIGN MR

Table 3.3-1

Basis for Conceptual Designs

REFERENCE ENGINE

The reference engine is the AJ10-221 thruster developed by us on NAS3-25646. It has a nominal thrust of 490 Newtons at a chamber pressure of 115 psia, using NTO/ MMH propellants at MR 1.65. With a 286:1 nozzle it has a demonstrated performance of 321.8 sec. The chamber is Ir-lined Re with a fuel-cooled front end, a 92-element platelet injector, and a silicide-coated C-103 skirt. The chamber has been fired for over 6 hours without sign of damage. The development of this engine technology has been described in Refs.9, 10, and 11.

CONCEPT 1A

This engine is directly scaled from the AJ10-221. It uses the same basic injector, with design improvements to lower its fabrication and test costs, and changes in hydraulic balance to optimize it for minimum pressure drop operation, with NTO/hydrazine, at much higher chamber pressure.

The throat diameter has been reduced to give 250 psia Pc at 100 lbf. To provide stability at low injector delta P, chamber volume has not been reduced. The cooled front end has been modified by adding a thermal barrier of plasma-sprayed zirconia, which reduces the heat transfer to the fuel and provides improved thermal margin.

Increasing the chamber pressure will result in an engine envelope length reduction of about 30% (9 in.) and a diameter reduction of 33% (4.5 in.). Because of the pressure increase and propellant change, performance has increased by 2.5% (8 sec).

The new design will use low-cost solenoid valves. It will require low-cost fabrication and assembly techniques for the Ir-Re chamber and C-103 skirt to be competitive. To assure adequate stress margins, the throat section will be increased locally from the 0.07 in. of the AJ10-221 to 0.25 in.

CONCEPT 2A

We have recently received a contract to build and test an engine design previously prepared for the SDIO Brilliant Pebbles project. The nominal operating point of this thruster is 100 lbf at a chamber pressure of 500 psia, using NTO/hydrazine, which exactly matches the requirements of case 2A. This engine will be built and tested during the next 12 months, so its status will be well demonstrated before flight engine design begins in Option 3. Therefore, the Case 2A concept is based on the use of the Brilliant Pebbles divert thruster injector.

This concept is assumed to be pump-fed to enable it to achieve the full advantages of the 500 psia chamber pressure. It is projected to have a delivered specific impulse of over 335 sec, about 4% increase over the reference engine. Because of the substantial reduction in chamber diameter, the chamber heat transfer is nearly 7 times that of the reference engine. For this reason, the chamber will use a Pt-Rh film cooled trip for front end thermal management. The chamber diameter will be about 0.12 in larger than Brilliant Pebbles to accommodate the trip without excessive pressure drop which otherwise would occur at the high subsonic chamber Mach number.

The Brilliant Pebbles injector is a titanium platelet design with splashplate elements. The BP carbon composite chamber will be replaced with the Ir-Re chamber to provide the much longer life required for our applications. The nozzle skirt can be either C-103 or carbon composite.

Some of the considerations which guided these concept designs from the standpoint of performance, heat transfer, and stability are discussed in the following Sections.

3.3.1 Performance Determination

Predicted Performance

Performance predictions for different engine designs, propellants and operating conditions are calculated using JANNAF methodology. The results of these calculations for the two engine concepts are shown in Table 3.3.1-1.

The performance prediction procedure can be followed stepwise down the table. First, theoretical performance is calculated for the propellant combination of interest, as a function of mixture ratio, area ratio, and chamber pressure, assuming one-dimensional isentropic expansion of the combustion products, which remain in chemical equilibrium. This produces the ODE performance. This value, and the results of subsequent calculation steps, are illustrated in Figure 3.3.1-1, which compares results for MMH and hydrazine at low and high pressure.

Next, a more realistic gas composition is calculated, using finite reaction rate kinetics and the performance is re-determined. Because reactions such as $H+H=H_2$ and $O+CO=CO_2$ do not go to completion, less potential energy is available for conversion to kinetic energy in the nozzle, and therefore the ODK performance is lower than ODE. The effect of chamber pressure and mixture ratio on kinetics efficiency is illustrated in Figure 3.3.1-2.

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PREDICTED PERFORMANCE FOR HIGH-PRESSURE EARTH-STORABLE THRUSTER CONCEPTS

PARAMETER	VALUES		COMMENTS
CASE	1A	2A	Concept designs
CHAMBER PRESSURE, PSIA	250	500	Operating chamber pressure
MR, O/F	1.15	1.15	Mixture ratio (optimum to be determined in Tasks 2 and 4)
AREA RATIO, Ae/At	300	300	Nozzle area ratio; can be modified to meet specific envelope/performance needs
Rthroat, INCHES	0.26	0.185	Throat radius
Lnozzle, INCHES	13.2	9.4	Nozzle length, throat to exit
% BELL	83.4	83.4	Percent of length of 30 deg conical nozzle to same exit diameter
Isp ODE, LBF-SEC/LBM	357.3	357.5	Theoretical performance based on 1-dim. equilibrium composition
Isp ODK, LBF-SEC/LBM	349.6	352.3	Theoretical performance based on 1-dim. kinetics-limited composition
Isp TDE, LBF-SEC/LBM	353.0	353.1	Theoretical performance based on 2-dim. equilibrium composition
Isp bl, LBF-SEC/LBM	7.4	6.2	Performance loss due to boundary layer
eta kinetics, %	97.8	98.6	Kinetics efficiency
eta divergence, %	98.8	98.8	Divergence efficiency
Isp, PI, LBF-SEC/LBM	337.9	341.8	Performance of a perfect injector
ERE, %	98	98	Combustor energy release efficiency
Isp del, LBF-SEC/LBM	331	335	Predicted delivered performance
THRUSTER EFFICIENCY, %	92.6	93.7	Overall thruster efficiency, based on ODE theoretical performance

Table 3.3.1-1

Predicted Performance for High-Pressure
Earth-Storable Thruster Concepts

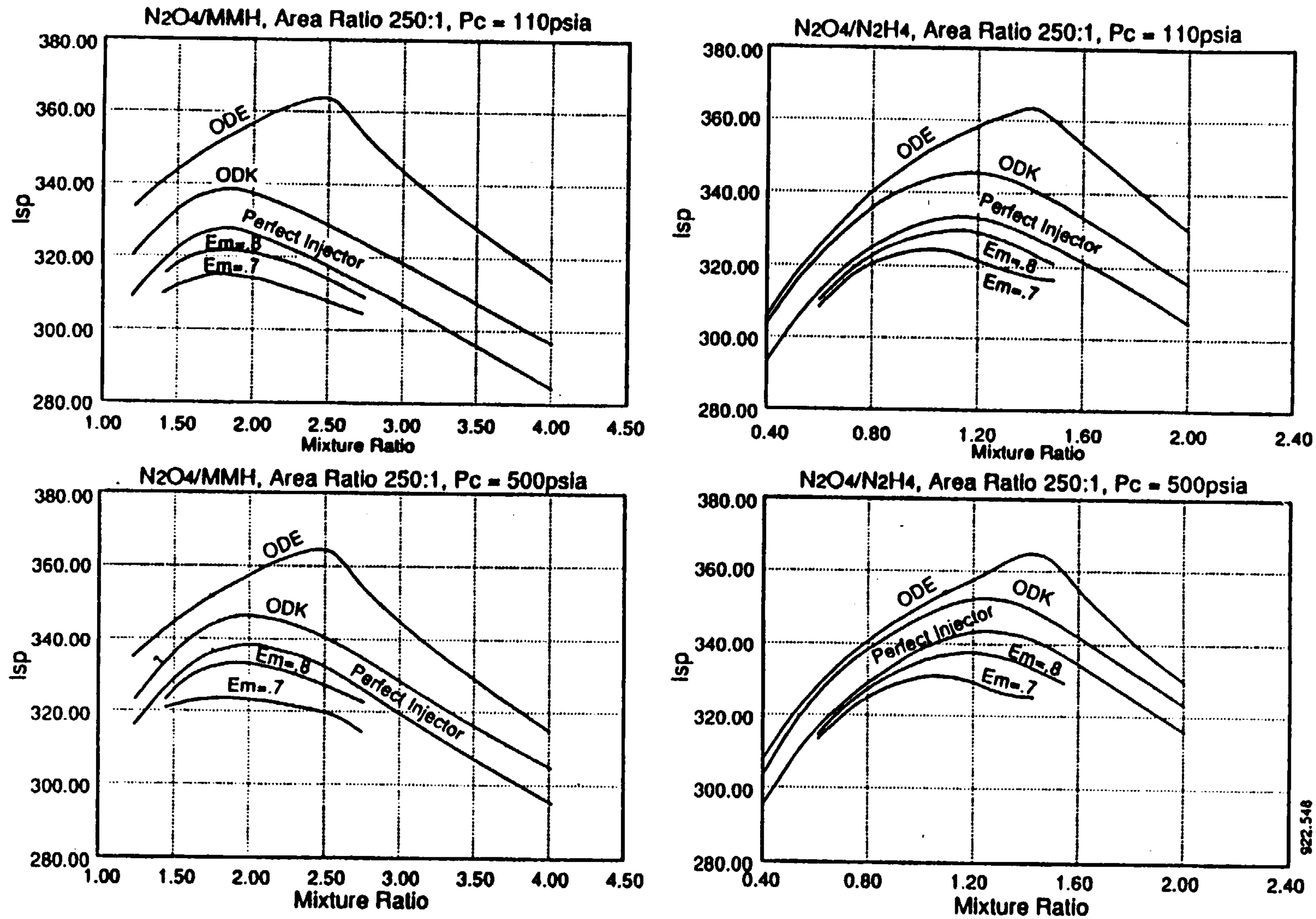

Figure 3.3.1-1. Performance Losses Are Dependent on Propellant and Chamber Pressure Performance

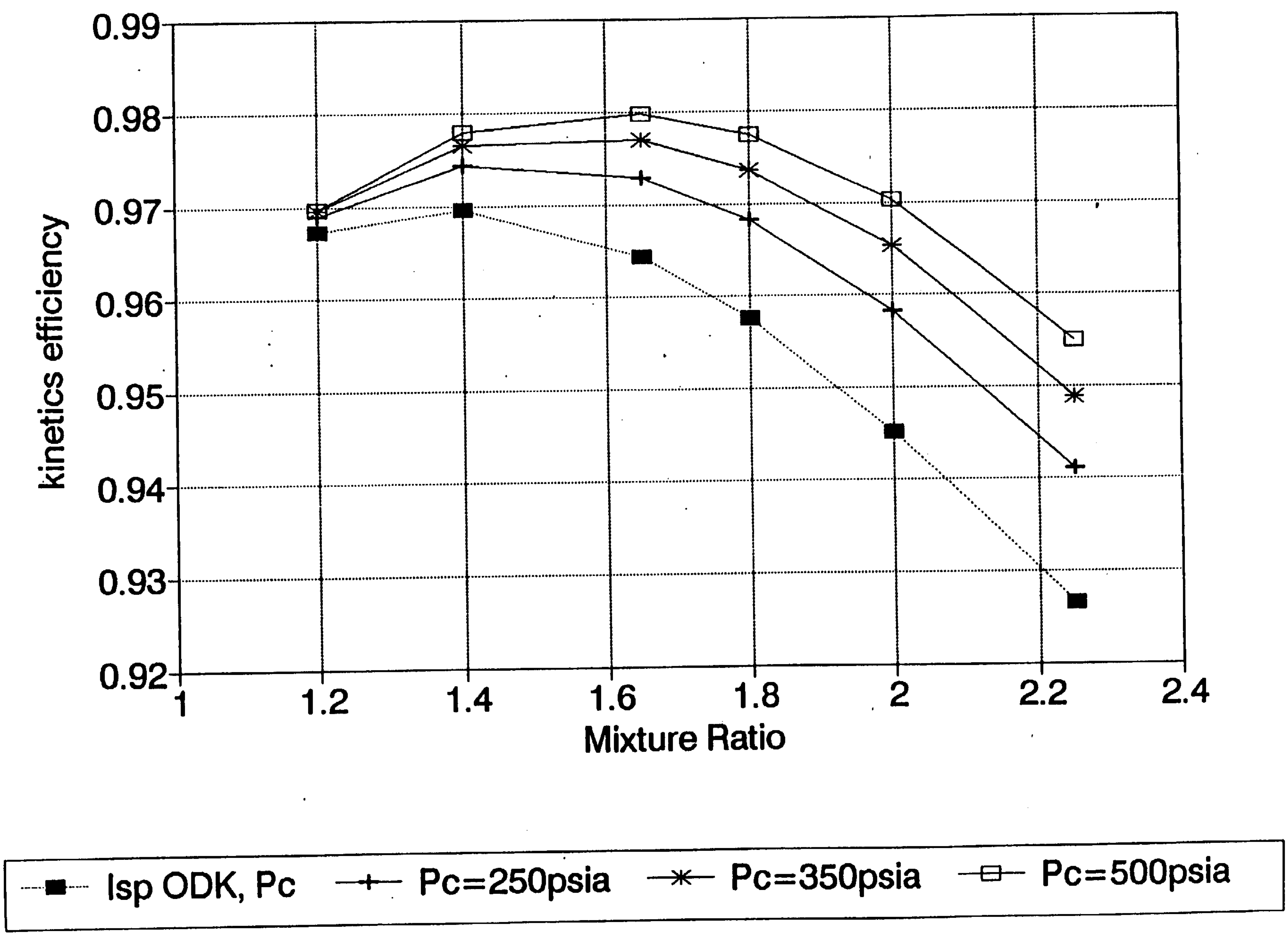

Figure 3.3.1-2. N_2O_4/MMH, 250:1 Area Ratio, 100 lbf Vac

The "perfect" injector performance is then calculated. The ODK performance is reduced to account for boundary layer losses, (thermal and friction), and nozzle divergence losses, since the optimum nozzle design normally does not produce parallel flow at the exit. The effect of increasing chamber pressure is to reduce boundary layer loss, as illustrated in Figure 3.3.1-3. The perfect injector performance must then be penalized for the effects of mixing losses caused by vaporization delay and imperfect mixing and reaction of the propellants. This latter term is significant for thrusters which purposely employ non-uniformity in the form of fuel-film cooling (FFC) to lower the chamber wall temperature. The effect of the mixing loss on performance is shown in Figure 3.3.1-4, for MMH and hydrazine, for a range of 0 to 20% FFC.

Accounting for the energy release efficiency (ERE) gives the expected delivered performance of the thruster. This process gives a performance prediction which is in close agreement with measured test data.

Experimental Performance

We will determine thruster performance in this program by measuring thrust directly and correcting for measured ambient pressure. The correction term is small, well understood, and straightforward to apply.

An alternate approach which is employed is to determine performance from measurement of chamber pressure and use of a thrust coefficient for the nozzle. These two approaches are compared in Table 3.3.1-2.

Both approaches require accurate measurement of propellant flow rate. The direct measurement approach requires accurate measurement of thrust, which is a technology we have in hand, and follows the JANNAF performance determination procedures. It permits comparison of thruster performance between test facilities on the same basis. The C*-Cf approach requires knowledge of several parameters, Pc, At, and Cf which are not easy to measure or calculate and is therefore subject to large uncertainty.

Combustion Efficiency

A consistent definition of combustion efficiency can be derived by combining the predicted and measured vacuum specific impulse. For example "thruster efficiency" can be defined by the ratio of (Measured Is)/(ODE Is)*100; this parameter is shown on Table 3.3.1-1 for the three thrusters.

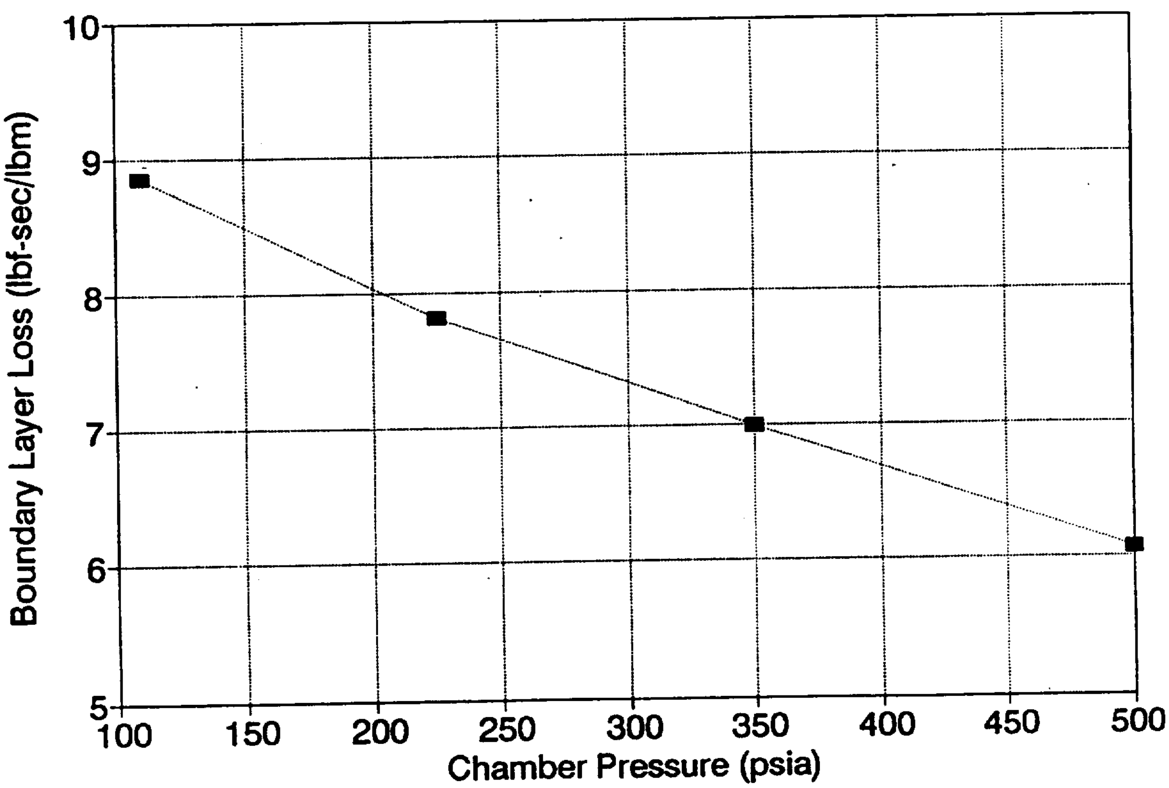

Figure 3.3.1-3. **N_2O_4/MMH, 100 lbf Vac, 250:1 Area Ratio, B.L. Loss Decreases With Increased Pc**

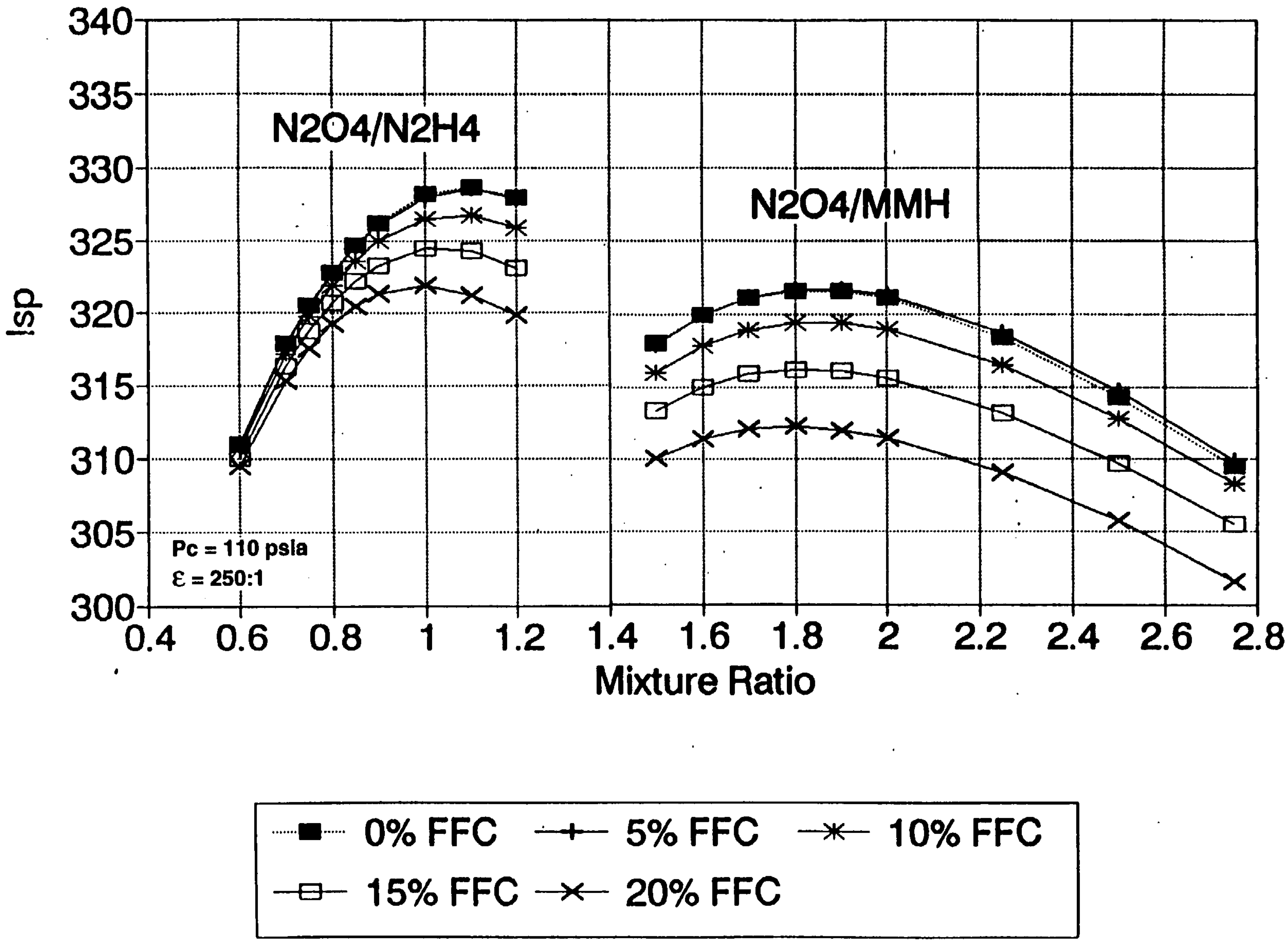

Figure 3.3.1-4. Delivered Isp With Em = 0.8 and FFC

Table 3.3.1-2

Comparison of Performance Calculation Methods

COMPARISON OF PERFORMANCE CALCULATION METHODS

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[UPDATED FROM HUGHES79]

NOMINAL CONDITIONS:	DESCRIPTION	DATA SOURCE	VALUES	COMMENT ON MEASUREMENT
BASIS:			TESTS AT 286:1	
AREA RATIO	Ae/At	DESIGN	286:1	
PROPELLANTS		DESIGN	NTO MON-3/MMH	
MIXTURE RATIO	Wo/Wf	DESIGN	1.65	
VAC. THRUST, LBF	Fv	DESIGN	110.2	
1. THRUST-BASED PERFORMANCE				
PROPELLANT FLOW, LBM/SEC	Wt	MEASUREMENT	0.3424	ACCURATE WHEN DONE WITH PROPELLANT-CALIBRATED FLOW METERS
MEASURED ALTITUDE THRUST, LBF	Fa	MEASUREMENT	108.1	ACCURATE IF WELL-ENGINEERED STAND IS USED
EXIT AREA, IN^2	Ae	MEASUREMENT	145.4	ACCURATE
AMBIENT PRESSURE, PSIA	Pa	MEASUREMENT	0.0145	ACCURATE IF SUITABLE RANGE TRANSDUCERS USED
VACUUM THRUST CORRECTION, LBF	Fc=Pa*Ae	CALCULATION	2.11	SMALL CORRECTION; WELL UNDERSTOOD
VACUUM THRUST, LBF	Fv=Fa+Fc	CALCULATION	110.2	BY DEFINITION
VACUUM SPECIFIC IMPULSE, LBF-SEC/LBM	Isv=Fv/Wt	CALCULATION	321.8	BY DEFINITION
2. Cf-BASED PERFORMANCE				
PROPELLANT FLOW, LBM/SEC	Wt	MEASUREMENT	0.3424	ACCURATE WHEN DONE WITH PROPELLANT-CALIBRATED FLOW METERS
WALL STATIC PRESSURE, PSIA	Ps	MEASUREMENT	114.8	ACCURATE
RAYLEIGH LINE CORRECTION, PSIA	Pr=f(COMBUSTION, Mn)	CALCULATION	0.75	CORRECTION ~2%
MACH NUMBER CORRECTION, PSIA	Pm=f(Mn)	CALCULATION	0.75	CORRECTION ~2%
TOTAL PRESSURE AT THROAT, PSIA	Pc=Ps-Pr+Pm	CALCULATION	114.8	BY DEFINITION
COLD THROAT AREA, IN^2	At	MEASUREMENT	0.5115	ACCURATE
THROAT TEMPERATURE, oF	Tt	MEASUREMENT	3300	ACCURATE IF MEASURED
COEF THERMAL EXPANSION, IN/IN-oF	Ct	MEASUREMENT	4.17E-06	DATA SOURCESFOR MOST MATLS HAVE SCATTER AT HIGH TEMP
HOT FIRE THROAT AREA, IN^2	Ath=At(1+Ct*Tt)^2	CALCULATION	0.5257	MODERATE UNCERTAINTY, ESP. WITH HOT THROAT
VACUUM THRUST COEFFICIENT	Cf=f(TDK) or prior meas.*	CALCULATION	1.826	COMPLEX ANALYSIS/PREVIOUS MEASUREMENT*
VACUUM THRUST, LBF	Fv=Cf*Pc*Ath	CALCULATION	110.2	BY DEFINITION
VACUUM SPECIFIC IMPULSE, LBF-SEC/LBM	Isv=Fv/Wt	CALCULATION	321.8	BY DEFINITION

***If Cf is determined from prior altitude thrust measurement, it is accurate only under the same conditions, i.e.: injector, chamber and nozzle configuration, mixture ratio and propellant flow.**

3.3.2 Heat Transfer Determination

Table 3.3.1-1 includes the results of thermal analysis conducted to compare the heat flux and chamber temperatures of the reference engine to the two concept thrusters.

Summary

A series of parametric thermal analyses were conducted to establish the relationship between operating pressure and the maximum chamber wall temperature for a radiation cooled 100 lbf thrust iridium lined rhenium chamber. The simplified two dimensional heat transfer model was calibrated by comparing the predicted maximum chamber temperatures with the extensive hot fire test data generated from existing iridium lined rhenium chambers tested with NTO/MMH propellants at 110 psia. Parameters investigated in this analysis are: chamber pressures up to 500 PSIA, a change of fuel from MMH to N_2H_4; and increasing the chamber wall thickness to withstand the higher operating pressures and temperatures. A design limit of 4000°F was placed on the chamber material.

The model was successfully able to match the test data using existing Aerojet analytical methods for low thrust engines, [Ref. 12]. The analyses indicated that chamber pressures of up to 500 psia could be employed, without exceeding the imposed 4000°F, without the use of fuel film cooling. Increasing the chamber wall thickness was found to be effective in reducing the maximum temperatures as a result of the two dimensional heat conduction effects in the throat region.

Method of Analysis

Gas Properties

The combustion temperature and thermodynamic and transport properties; specific heat, viscosity, thermal conductivity and Prandtl number were computed from the standard JANNAF codes as a function of chamber pressure and mixture ratio. The selection of the appropriate property states; i.e. equilibrium or frozen, for the analysis, was based on previous calibration experience, as shown in Figure 3.3.2.-1 and Table 3.3.2-1.

During calculation of performance parameters, heat transfer design data are also calculated. For example, combustion temperature for NTO/MMH and NTO/hydrazine is plotted as a function of chamber pressure and mixture ratio in Figure 3.3.2-2.

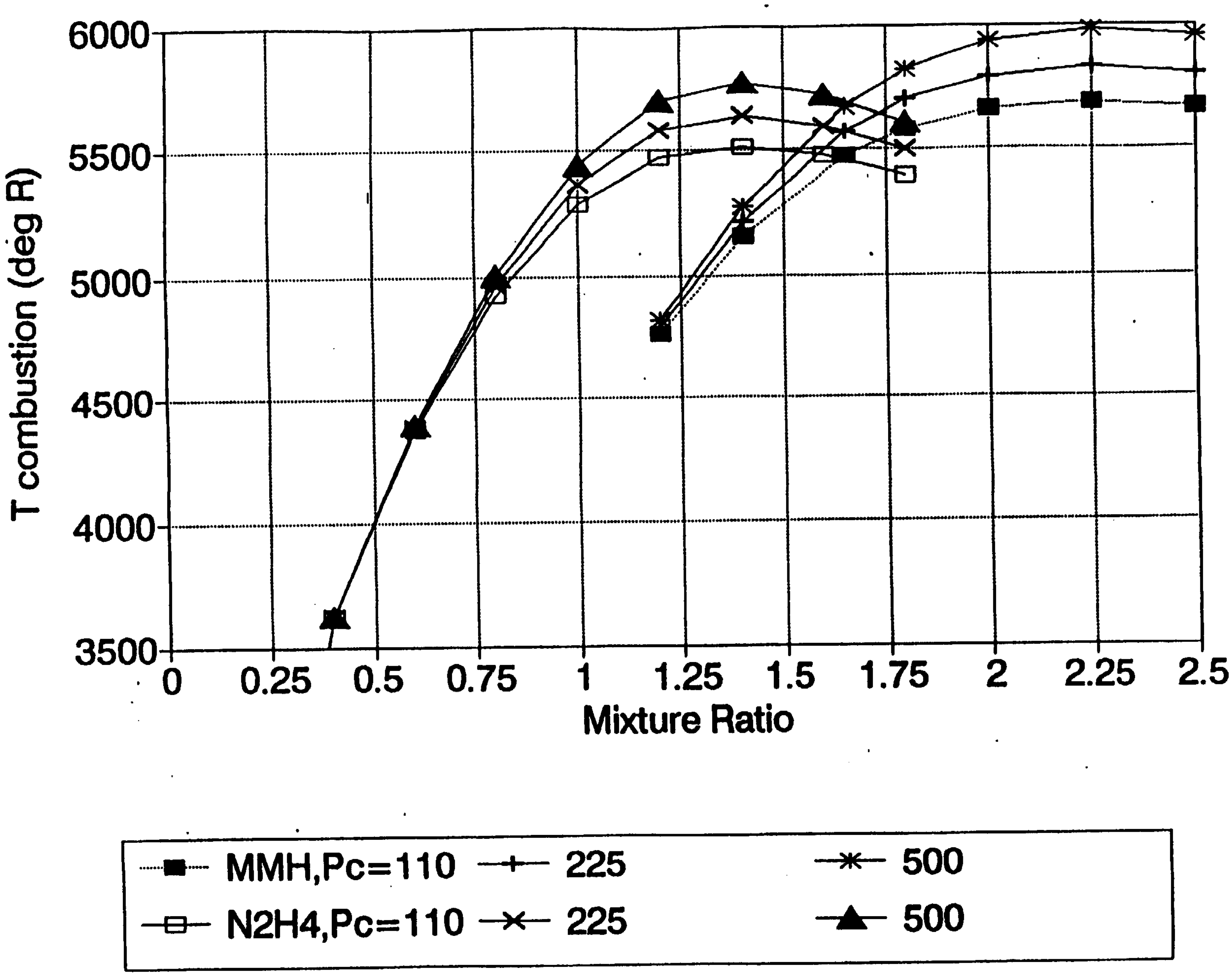
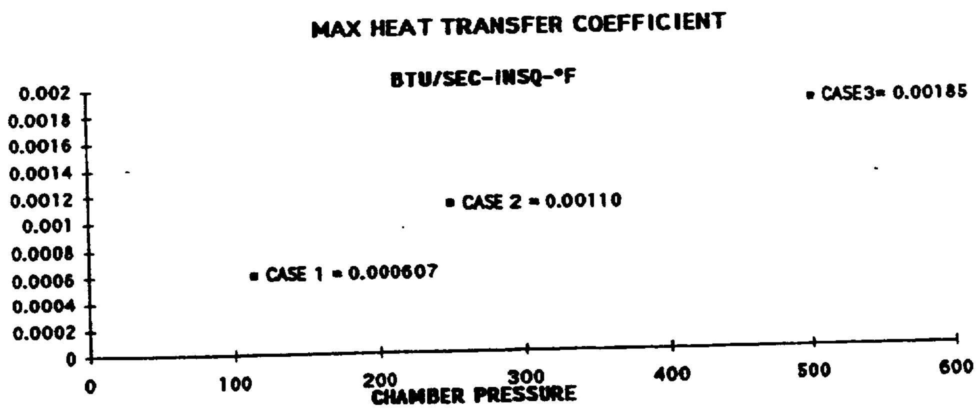


Figure 3.3.2-1. ODE Combustion Temperature

NTO/MMH					
MR	THROAT T°R	CP	MU	K	PR
1.65	5191	0.508	4.78E-06	4.15E-06	0.584
1.80	5358	0.499	4.96E-06	4.14E-06	0.599
	CHAMBER T°R				
1.65	5572	0.512	5.03E-06	4.45E-06	0.579
1.80	5702	0.503	5.19E-06	4.40E-06	0.593
NTO/N2H4					
MR	THROAT T°R	CP	MU	K	PR
0.80	4507	0.574	4.16E-06	4.34E-06	0.55
1.00	4969	0.554	4.64E-06	4.43E-06	0.584
1.20	5250	0.522	4.97E-06	4.31E-06	0.614
	CHAMBER				
0.80	4967	0.583	4.48*10^-6	4.77*10^-6	0.548
1.00	5369	0.56	4.92*10^-6	4.74*10^-6	0.58
1.20	5583	0.538	5.19*10^-6	4.60*10^-6	0.607

Table 3.3.2-1. Gas Heat Transfer Properties



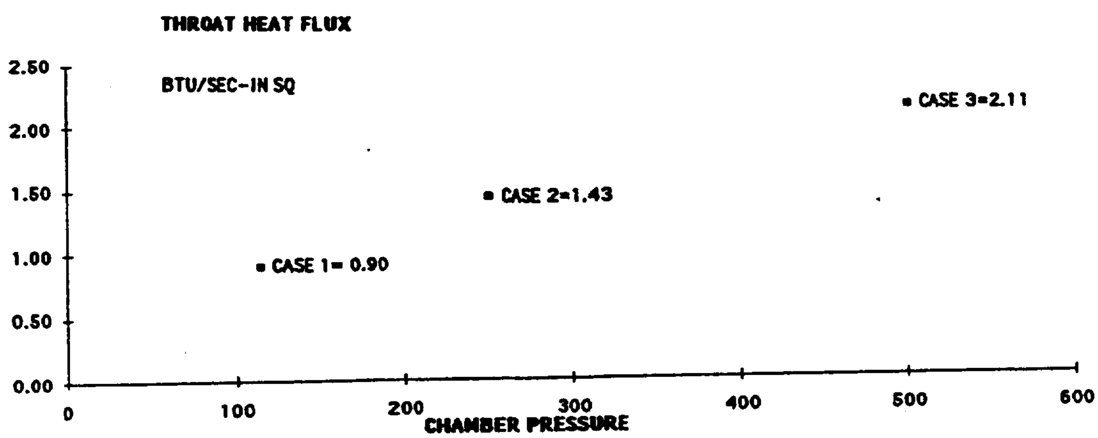


Figure 3.3.2-2. Thruster Heat Transfer vs. Chamber Pressure

Convective Boundary Conditions

The convective thermal boundary conditions at the maximum wall temperature location, a short distance up stream of the throat, were computed using the simplified laminar boundary layer procedure of Ref. 6. This included the application of the recommended 1.3 multiplier to account for the strong pressure gradients and wall cooling effects in the near throat and throat region.

The thermal heat loads at the chamber head end and cylindrical region were also computed by the method of Ref. 6 using the prescribed method of Bartz for the turbulent combustion region where the injector design strongly influences the heat transfer rates. An experimental heat transfer enhancement factor of 1.24 was applied for the splash plate injector. This value was obtained from the experimental data of Refs. 13 and 14.

Wall Conduction

The selected two dimensional heat conduction model represents the radial heat flow in a thick wall cylinder convectively heated on the inside, and radiation cooled on the outside. Axial heat conduction along the length of the chamber was not considered as being of great enough significance at this time to justify the additional analysis costs. The thermal conductivity of rhenium was taken from published data including supplier and handbook values. Differences of more than 20% could found between references, none of which were for the specific crystal form of CVD rhenium. An average value of 0.000607 Btu/(sec.-in -°F) for wrought material, at 4000°F, was employed for the analysis although it is suspected that the high purity and columnar structure of the CVD material make this a conservative selection. The thin iridium liner was assumed to have the same thermal properties as the rhenium.

Radiation

Radiation from the dentoid external surface of CVD rhenium was based on the experimental calibrations of Ref. 13 which indicated that the surface behaves as a black body. A view factor of 0.9 was found to provide an excellent calibration with the test data.

Calibration

A base line calibration analysis of the 110 lbf iridium lined rhenium engine operating at 115 psia at a mixture ratio of 1.65, case 1 of the thermal design studies, resulted in a predicted wall temperature of 3380°F which is about 20 °F higher than the measured values for these test conditions, Ref. 14.

The chamber region heat transfer coefficients and resulting heat flux exactly matched the test results, based on the fuel regenerative coolant temperature rise, Ref. 14, when the 1.24 factor is applied as discussed above.

Results

Two new design configurations were evaluated for the new high-performance, high-pressure engines using N_2H_4 as the fuel.

The first new configuration, case 2, has a maximum operating pressure of 250 psia, a mixture ratio of 1.15, and delivers an Isp of 330 sec. at an area ratio of 300:1. The throat diameter for this configuration is 0.521 in. and the chamber diameter is held at 1.7 in. which is the same as the reference 115 psia design.

The predicted maximum wall temperatures, as a function of wall thickness are given in Table 3.3.2.-2.

Table 3.3.2-2.

Chamber Maximum Wall Temperature vs
Thickness At 250 PSIA, MR= 1.5

Wall Thickness in.	Maximum Chamber Temperature °F
0.104	3860
0.156	3830
0.262	3790
0.521	3700

The maximum temperature is noted to be slightly sensitive to wall thickness. The two dimensional heat conduction within the thicker wall provides a slight benefit in reducing the maximum temperature value. Note the wall thicknesses of the previously tested chambers ranged between 0.060 and 0.070 in.

The chamber region heat flux to the fuel regeneratively cooled region was calculated for two configurations; one with a cool metal wall at 500 °F, and a second with a thermal barrier coating assumed to operate at a surface temperature of 2500°F. The chamber

CASE	MR	W	PC	TO	DTI	REY NO	ST*PR^.67	THROAT HG	TR °R
MMH	REF 1.65	0.342	115	5500	0.804	1.13E+05	0.00095042	0.00046658	5328.90428
N2H4	1.15	0.303	250	5500	0.521	1.49E+05	0.00082882	0.00085301	5445.89538
N2H4	1.15	0.299	500	5600	0.368	2.08E+05	0.00070122	0.00142743	5524.25353

CASE	CH PRESS	THROAT HG	THROAT FLUX	CH FLUX 500°F	CH FLUX 2500°F
1	115	6.07E-04	0.90	1.78	1.01
2	250	1.11E-03	1.43	1.66	0.94
3	500	1.86E-03	2.11	12.10	6.89

W = Flow Rate, lb-sec
TO = Combustion Temperature, °R
PC = Chamber Pressure, psia
DTI = Throat Dia, ID in.
TWI = Throat Wall Temperature, °F
Hg = Heat Transfer Coefficient, BTU/sec-in.^{2} °F
TR = Recovery Temperature, °R
Rey = Reynolds Number
TBC = Thermal Barrier Coating Surface Temperature, °F

Table 3.3.2-3. Chamber and Throat Heat Transfer Parameters

region heat flux and the advantage of the thermal barrier coating in reducing the heat flux is given in Table 3.3.2-3

The second new configuration case 3, has the same thrust and mixture ratio as the first higher pressure design, but is designed to operate at a chamber pressure of 500 psia. This design will deliver a specific impulse of 335 sec. This results in a still smaller throat diameter, 0.368 in. The chamber diameter has been reduced from 1.7 in. to 0.57 to miniaturize the engine. The relation between wall thickness and maximum temperature is given in Table 3.3.2-4.

Table 3.3.2-4

Chamber Maximum Wall Temperature vs Thickness
At 500 PSIA, MR= 1.15

Wall Thickness in.	Maximum Chamber Temperature °F
0.184	4078
0.368	3997
0.551	3950
0.736	3926

The optimum thermal design for this pressure will provide a wall thickness at the throat of between 0.3 and 0.5 in. Thicker values can be used but will probably result in excess weight and cost for this material.

The calculated chamber region heat transfer parameters for case 3 are given in Table 3.3.2-3. The reduction in chamber diameter has resulted in a much higher heat flux, even with the addition of the thermal barrier. It is therefore unlikely that the regeneratively cooled head end design approach can be employed. The use of a highly fuel film cooled head end design with subsequent elimination of the coolant using the patented "Two Stage Combustor" design represents one potential approach to eliminating the cooling associated loss in performance. Another is to use a larger chamber diameter, i.e. as in case 2.

Recommendations

The initial heat transfer verification test program will be configured to verify these predictions along with the chamber wall chemical compatibility which is considered to be less predictable.

3.3.3 Stability Considerations

The reference thruster and the two flight concept designs incorporate splashplate elements that have been well characterized with regard to both high frequency and chug instability. Therefore, stability assessment simply requires examination of chamber pressure measurements obtained with Kistler and Taber pressure transducers. The high frequency Kistler pressure transducer will be close coupled to the chamber and monitored up to its frequency limit of about 25kHz since such high resonant frequencies are possible with 100 lbf-class thrusters.

The splashplate element used in the concept injectors is well-characterized from a combustion stability standpoint. Table 3.3.3-1 shows some of our stability experience with injectors in the 0.5 to 6000 lbf-class. The splashplate element exhibits an "injection coupling" mode of instability and, therefore, its stability characteristics are a function of its injection time lag, injection stiffness (delta P/Pc), and the acoustic resonance frequencies of the thrust chamber.

The splashplate element is ideally suited for our high-pressure earth-storable concepts because we can change its injection response and stiffness by appropriate changes in nozzle throat size and thrust level (that is, flowrate and, therefore, injection velocity and delta P/Pc ratio).

This relationship is shown for the proposed engines in Figure 3.3.3-1. The shaded zone shows the chamber resonant frequencies and response for chug, 1L, 1T and 2T acoustic modes for case 1A (approximately 5400, 14,000, and 23,300 Hz, respectively). Basically, an engine operating curve which intersects the shaded zone could operate unstably at the indicated resonance, with a magnitude of instability which depends on system damping.

Our standard injector/chamber design uses an acoustic resonator tuned for the first tangential mode to provide additional stability margin through added damping for this mode. The upper curve shows the approximate range of test bed operation when throttled from an initial condition of Pc=100 psi, F=150 lbf. The lower curves show the operation with reduced throat size/higher Pc. At a chamber pressure of 100 psia the engine has very stiff propellant injection

Table 3.3.3-1

Stability Experience With Aerojet
Platelet Injectors

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STABILITY EXPERIENCE WITH AEROJET PLATELET INJECTORS

ENGINE	BASIS	CHAMBER DIA., IN	THROAT DIA, IN	THRUST, LBF	CHAMBER PRESSURE, PSIA	INJECTOR DELTA P PSI	DELTA P/ Pc	COMMENT
HALF-POUND	MEAS.	23	0.6	0.5	120	300	2.50	
ALAS ACS	MEAS.	0.25		10	500	400	0.80	
5 LBF	MEAS.	0.35	0.16	5	130	60	0.46	
14 LBF	MEAS.	0.65	0.32	14	100	100	1.00	Still chug stable at Pc=35 psia
BRILLIANT PEBBLES	CALC.	0.65		100	500	400	0.80	
LDI-2	MEAS.	0.86	0.65	300	550	500	0.91	OK to Pc=360
TACAWS	MEAS.	0.88	0.65	570	1200	600	0.50	
LDI-1	MEAS.	0.92	0.65	300	550	500	0.91	OK to Pc=360
MIB100	MEAS.	0.95		100	120	60	0.50	
SCALABLE INJ.	MEAS.	1.35		150	125	60	0.48	
AJ10-221	MEAS.	1.7	0.8	100	100	60	0.60	OK to Pc=50
XLR-132	MEAS.	2		3750	1500	500	0.33	Chug Pc<400
OME SUBSCALE	MEAS.	2.7		600	150	40	0.27	
870 LBF ACS	MEAS.	3.2		870	150	60	0.40	
MX	MEAS.	6		4000	175	70	0.40	
LCAE--2500	MEAS.	5.85	3.29	2500	150	75	0.50	Chug at Pc~115
LCAE--4000	MEAS.	5.85	3.29	4000	260	75	0.29	Ok @ Pc=170; 650 Hz chug @ Pc=150
TRANSTAR I	MEAS.	5.5		3750	350	140	0.40	
UPRATED OME	MEAS.	5.5		6000	350	140	0.40	1L mode Pc<175
OMS ME	MEAS.	8.11	5.85	6000	125	33	0.26	1L mode Pc<110

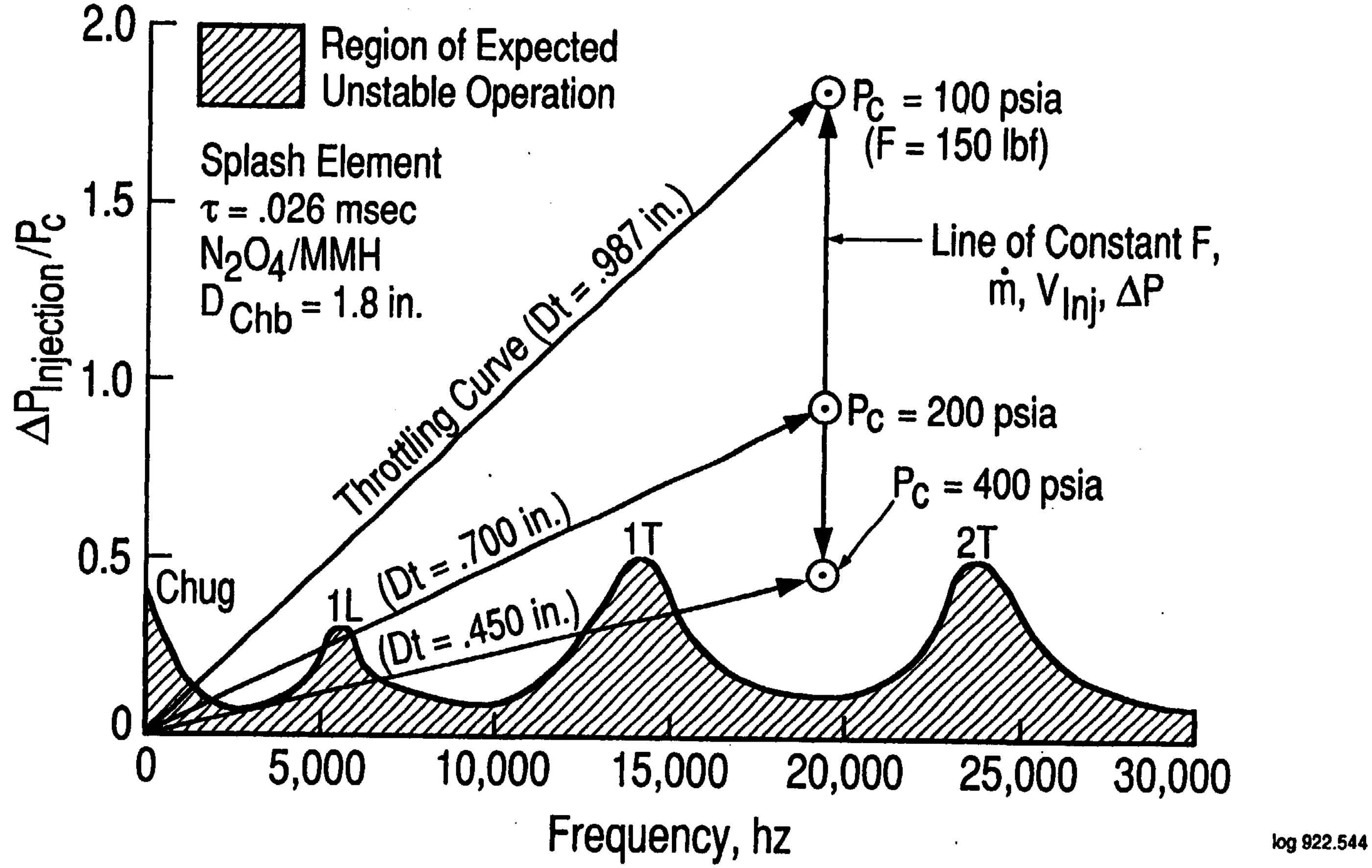

Figure 3.3.3-1. Injection Response Can Be Achieved by Throttling the Testbed Engine

(delta P/Pc=1.8) and therefore will provide stable operation. When the chamber pressure is increased through nozzle size reduction at a constant thrust level, the injection delta P-to-Pc ratio will decrease in proportion to the chamber pressure increase. Operating points at Pc= 100, 200, and 400 psia are noted on the figure; all are in the stable operating region. At the high Pc condition, the delta P/Pc is 0.2 and damping is required for the first tangential mode which occurs near 15kHz.

The concept 1A injector will be stiffened somewhat from the AJ10-221 design in conjunction with rebalancing for hydrazine; a compromise position must be taken to provide operation at low delta P while allowing an acceptable range of stable operation at off-design conditions. The results of stability calculations made for the reference engine and several concept engines are shown in Table 3.3.3-2 which gives the 1T and 1L values for these cases.

In our Task 2 and Task 4 rocket testbed testing we will measure engine stability and explore the complete operating range to determine if the design will be stable in the flight engine under its required range of operating conditions.

Table 3.3.3-2

Results of Stability Calculations for the Reference and Concept Engines

	AJ10-221	Concept 1A	Concept 1B (LDI)	Concept 2A (Brilliant Pebbles)
Thrust	100 lbf	100	100	100
Pc	115 psi	250	250	500
ε	286	300	300	300
D_c	1.71	1.71	0.92	0.65
D_{th}	0.81	0.52	0.52	0.37
CR	4	11	6.2	3.1
Isp	321	330	335	335
$\dot{\omega}_T$	.3115	.3030	.298	.298
ΔP/PC	.3	.3	.3	.3
ΔP (psid)	35	75	75	150
$V_{f\ inj}$ (fps)	72	105	105	148
$V_{o\ inj}$ (fps)	60	88	88	124
d_{comb} (in.)	.08	.08	.04	.04
τ_f (sec)	.000093	.0000635	.000032	.0000225
τ_{ox} (sec)	.000111	.000076	.000038	.000027
$freq_f$	5400	7875	15750	22200
$freq_{ox}$	4500	6600	13200	18600
Freq 1T (Hz)	15600	15600	29000	41000
Freq 1L	5428	5428	11400	11400
L' (in.)	4.2	4.2	2.0	2.0
N_e	92	92	162	
Propellants	NTO/Hydrazine All Cases			

4.0 RECOMMENDED TECHNOLOGY PROGRAM

SUMMARY

The recommended technology program is described in the Basic Contract Work Plan, Rev. 1.0, September 1993. This revision to the August Work Plan, submitted along with this Task 1 Report, recommends changing the nominal thrust level for the program from 22.5 lbf to 100 lbf and changing the fuel from monomethylhydrazine to hydrazine. To remain within the contract budget, the plan recommends some reduction in testing and elimination of procurement of spare testbed test hardware.

We believe that, given successful results during the Basic program, further revision to the plan for the Options would be feasible and beneficial. With positive results in hand from the Basic program, it should be possible to accelerate the pace of the Options while emphasizing the cost reduction aspects of the thruster technology, allowing earlier demonstration of flight-type rocket engines of a form suitable for user acceptance. These possibilities will be explored as the Basic Program progresses.

THRUST LEVEL CHANGE

As discussed in Section 3.2, detailed consideration of user requirements and technical limitations make the 100 lbf-class the area of highest possible user acceptance. A trend towards smaller spacecraft/lower axial thrust which we had believed to be developing is not supported by our most recent data sources. Where downsizing is evident (e.g., Iridium), the low total impulse eliminates the advantages of high pressure operation. There are secondary considerations: high performance ACS thrusters show no payoff for the ACS function but can result in reduced launch insurance costs since they would permit recovery from a delta-V engine failure with little loss in spacecraft on-orbit life. However, this is not a first-order benefit. The other evidence of thruster size reduction is from 200 lbf to 100 lbf.

The effects of the thrust level change on propellant requirements are shown in Table 4-1, which includes the effects of fuel change to hydrazine and recommended reductions in total firing time.

We have 100 lbf-class testbed hardware which will be used instead of the 14 lbf testbed. It will require modification for the Task 2 tests to include provision for interchangeable throats. At the 100 lbf thrust level our first preference for front end design is fuel regeneratively cooled. Again, we have a 100 lbf testbed cooled trip. To assure safe thermal management with

PROPELLANT UTILIZATION
REV. 1.0 AND N/C WORK PLANS

PROGRAM	NO-CHANGE WK PLN		REV. 1.0 WK PLN	
	NTO lbm	MMH lbm	NTO lbm	AH lbm
BASIC	185	96	231	192
OPTION 1	66	33	106	85
OPTION 2	991	498	833	705
OPTION 3	811	421	586	467
TOTALS, lbm	2,053	1,048	1,756	1,449

Table 4.1

Propellant Utilization Rev. 1.0 and
N/C Work Plans

hydrazine, we will grind out and reweld the critical inner weld. In addition, we will plasma coat this part with zirconia to reduce the heat transfer to the fuel.

FUEL CHANGE

The recommendation to change from MMH to hydrazine is driven by the need to maximize performance to be competitive with advanced systems now being proposed or developed. As discussed in Section 3.2.3 hydrazine has definite system advantages when used for spacecraft propulsion, in addition to its higher performance. Table 4-1 includes the effects of fuel change.

Use of hydrazine rather than MMH requires more margin on the front end thermal management, since the hydrazine is not tolerant of over-heating. This entails some initial tests with water-cooling to verify adequate margin at the high pressures.

FUTURE TECHNOLOGY ACTIVITIES

Because of the emphasis that NASA has placed on the downselect criteria for the Basic Program (Table 4-2), and the need to compare directly our performance in the specified tasks to TRW's, there is limited flexibility for this part of the program. We see that development and demonstration of alternate fabrication techniques and/or improved materials, at significantly lower cost than at present is essential to commercialization of this technology. Reliable, low-cost suppliers who have credibility with the spacecraft primes are also needed, as are second sources for the fabrication.

We believe that it should be possible to conduct some of the demonstration testing at Aerojet and TRW, in conjunction with the Basic Program, to reduce the time required to demonstrate viable technology for a high pressure rocket engine system.

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6-4-93

DOWNSELECTION CRITERIA
HIGH Pc PROPOSAL

[Pg J-11 OF RFP]

CRITERIA	POINTS	DESCRIPTION
1. RELEVANT DATA	300	THE AMOUNT, RANGE, AND QUALITY OF RELEVANT DATA GATHERED ON THE EFFECT OF HIGH PRESSURE ON THE COMBUSTION EFFICIENCY AND HEAT TRANSFER
2. INJECTOR/CHAMBER PERFORMANCE	250	THE BEST PERFORMING INJECTOR/CHAMBER CONCEPT WHICH IS JUDGED BY THE GOVERNMENT TO BE ABLE TO OPERATE WITHIN THE PROJECTED THERMAL LIMITS OF THE SELECTED CHAMBER MATERIALS
3. REALISTIC COSTS	250	THE EVALUATED, REALISTIC COSTS TO NASA LEWIS OF TESTING AND HARDWARE, AS JUDGED BY THE GOVERNMENT.
4. APPLICATION SELECTION	200	THE SELECTION OF APPLICATIONS AND ROCKET OPERATING ENVELOPE WHICH IS JUDGED BY THE GOVERNMENT TO HAVE THE MOST LIKELY CHANCE OF USER ACCEPTANCE.
TOTAL POINTS=	1000	

Table 4.2. Downselection Criteria High Pc Proposal

5.0 REFERENCES

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13. Schoenman, L., Franklin, J. and Lansaw, P.T., "Feasibility Demonstration of a High-Performance 100 lbf Rocket Engine," JPL Contract 957882, Final Report, Jan. 1989.

14. "Advanced Small Rocket Chambers," NAS 3-25646, Monthly Technical Progress Reports.

APPENDIX B

PRESENTATION OF TASK 2 ROCKET TESTBED DESIGN AND SUPPORTING DATA, 30 MARCH 1994

High Pressure Earth Storable Rocket Technology Program

Oral Presentation of Task 2 Rocket Testbed Design and Supporting Data

30 March 1994

NASA Lewis Research Center

17.130-4/1

High Pressure Earth Storable Rocket Technology Program

Agenda

08:00	Introduction/Meeting Goals/Executive Summary
	Exploratory Test Results Supporting Design
	Design of Task 4 Testbed
	Analysis Supporting Design
	Lunch
	Task 4 Test Plan
	Task 3 Fabrication
	Summarize; Action Items

17.130-4/2

HIPC PROGRAM PERSONNEL

		Phone (916) 355-
Cheri Cotter	Performance Analysis	6751
Stan Hart	Project Engineer	3639
Ross Hewitt	Stability, Injector Design	2664
Harvey Howard	Data Analysis	4858
Don Jassowski	Principal Investigator	2849
Richard Matthew-Rogers	Test Engineer	3700
Carolyne Montgomery	Contracts	2840
Shirley Reed	Reports	2340
Bryce Reimer	Program Manager	2177
Sandy Rosenberg	Fellow/Consultant	
Laura Ross	Fiscal	5191
Len Schoenman	Consultant	2964

High Pressure Earth Storable Rocket Technology Program

Program Objectives to Date

- Determine What the User Community Will Require for Future Satellite Propulsion – In Terms of Thrust, Propellants, Performance. etc.
- Using That Data, Perform a Test Program to Conceptually Verify the Effect of Different Chamber Pressures Using the Selected Thrust Level and Propellant Combination
- Design, Fabricate and Test Hardware to Prove the Result of the Conceptual Testing

17.130-4b/rt/8

High Pressure Earth Storable Rocket Technology Program

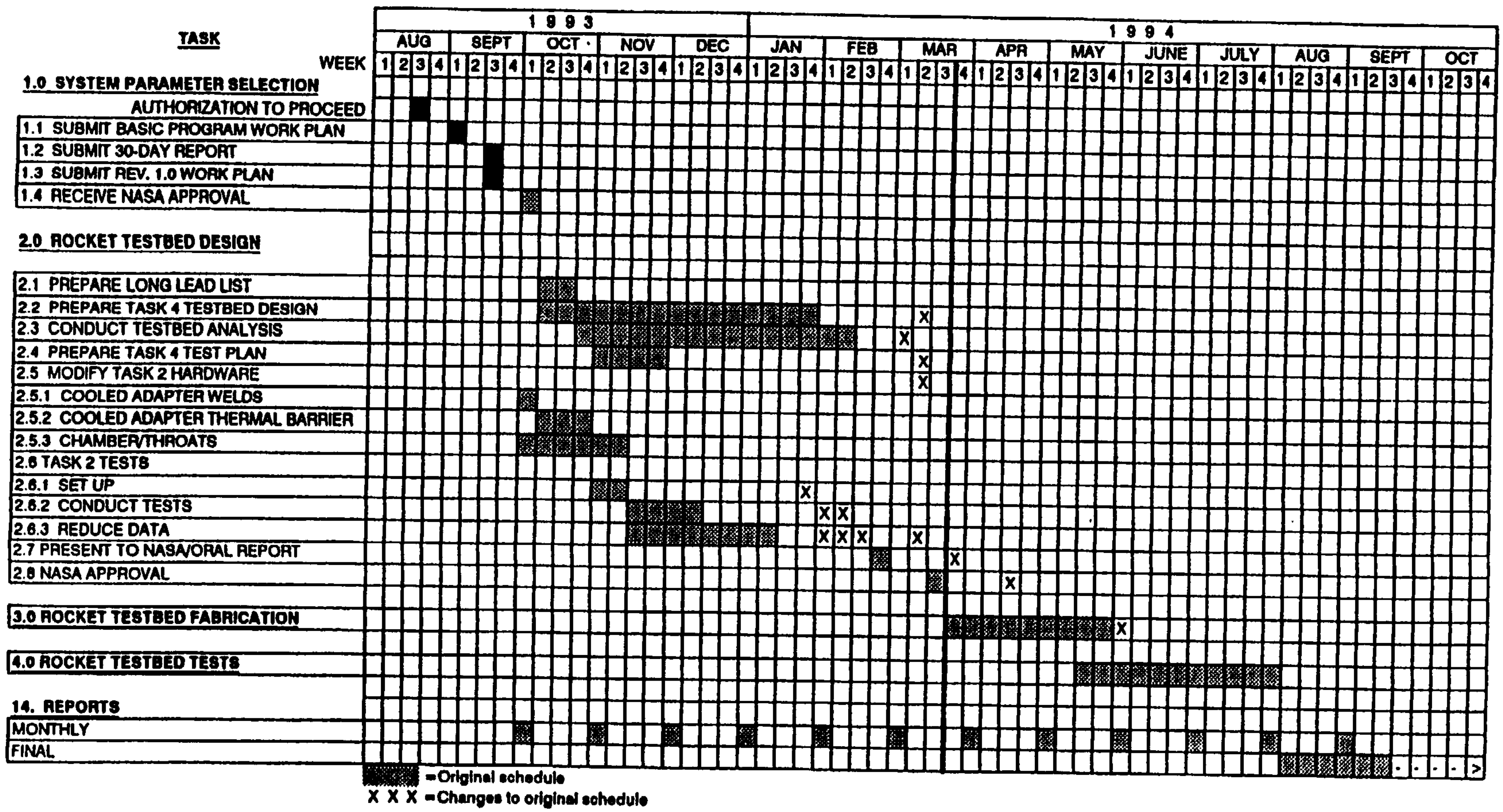
User Requirements and Desires

- User Survey and Review of Current Missions
  - High Performance is Most Important Factor
  - A 100 lbf Engine Was of More Use Than a 14 lbf Engine
  - Users Strongly Favor Pressure Fed Over Pump Fed From a Risk/Reliability Standpoint
  - Cost is an Important Factor; However a 25% Cost Increase Over Normal Is Acceptable for an Isp of At Least 330 Seconds

17.130-4b/rv9

High Pressure Earth Storable Rocket Technology Program

Program Is Still on Schedule



High Pressure Earth Storable Rocket Technology Program

Budget Has Changed But Still Adequate, $K

	Budget	Actual	Estimate at Completion	Estimate Variance
System Parameter Selection	28	25	25	+3
Testbed Design	158	153	158	0
Hardware Fab for Task 4	43	0	95	-52
Testbed Testing	151	0	114	+37
Administration	122	47	122	0
Management Reserve	12	0	0*	+12
Total Cost	514	225	514	0

- Change Due to 100 lbf Thruster Rather Than 15 lbf Thrust
- The "Testbed Design" (Task 2) Testing Used About $115K of Task 2 Budget
- Shortfall in FY 1994 Funding Will Move Task 4 Testing Analysis and Final Report Into FY 1995

17.130-4b/rt/14

High Pressure Earth Storable Rocket Technology Program

Concept Testing (Task 2)

- Examined Two Approaches, (1) Regenerative Cooled Head End and (2) Film Cooled With Higher Trip
  - Regen Cooled Head End
    - Regen Cooled Head End
      - Stable and Good Combustion Efficiency
      - Regen Portion Would Have to Be Redesigned for Use With Hydrazine
    - Film Cooled With Higher Trip
      - Test Duration Limited Due to Deterioration of Stainless Trip
      - Trip Height, Length and Material for Task 4 Testing Has Been Determined Based on Task 2 Test Results

17.130-4b/rt/10

High Pressure Earth Storable Rocket Technology Program

Summary of Results and Conclusions

- Significant performance increase results when Pc is increased from 100 to 250 psia (8 sec)
- Thermal management design established for Task 4 testbed
- Design for trip provides material compatibility; chamber compatibility to be determined in Task 4
- NTO/hydrazine proven stable under all operating conditions

Exploratory Test Results
Supporting Design of Task 4 Testbed

17.130-4/3

Task 2 Testing

The plan for the testbed was to use existing 100 lbf hardware, with the addition of free standing rhenium chambers for obtaining equilibrium thermal data. In early Task 2 exploratory testing the fuel regen cooled front end of this hardware proved to have inadequate thermal margin. Therefore, existing fuel film cooled testbed hardware was used to provide a firm basis for this approach for the Task 4 design.

An existing injector, S/N 5, was used for these tests, since it has provisions for film cooling. Its performance, but not its compatibility, had been characterized in the Advanced Small Rocket Chambers Contract. The injector proved to have a local oxidizer-rich spot which caused local over heating of the stainless trip and limited test duration in most tests. Except for local over heating the trip and sleeve operated at acceptable temperatures for stainless. For long duration hardware high temperature, oxidation resistant materials will be used with a S/N 7 injector to be built for the Task 4 tests.

Performance measurements indicated that operation at 250 psia chamber pressure gave the anticipated increase over 100 psia operation; operation at 500 psia did not show the expected increase, with the hardware employed. The low performance at 500 psi for the fuel film cooled case is believed to be caused by the decrease in trip mixing effectiveness at constant trip height, as chamber Mach number is decreased.

Heat transfer to the chamber was not expected to increase greatly since the tests were conducted at constant mass flux. However, the trip heat transfer was found to be about proportional to chamber pressure. The heat transfer at the trip, as determined by time required to reach 1500°F, correlated with PC ^ 0.8 MR ^ 1.5.

17.130-4a/rt/2

Task 2 Test Program Summary

17.130-4/4

High Pressure Earth Storable Rocket Technology Program

Summary of Testing Task 2 Testbeds

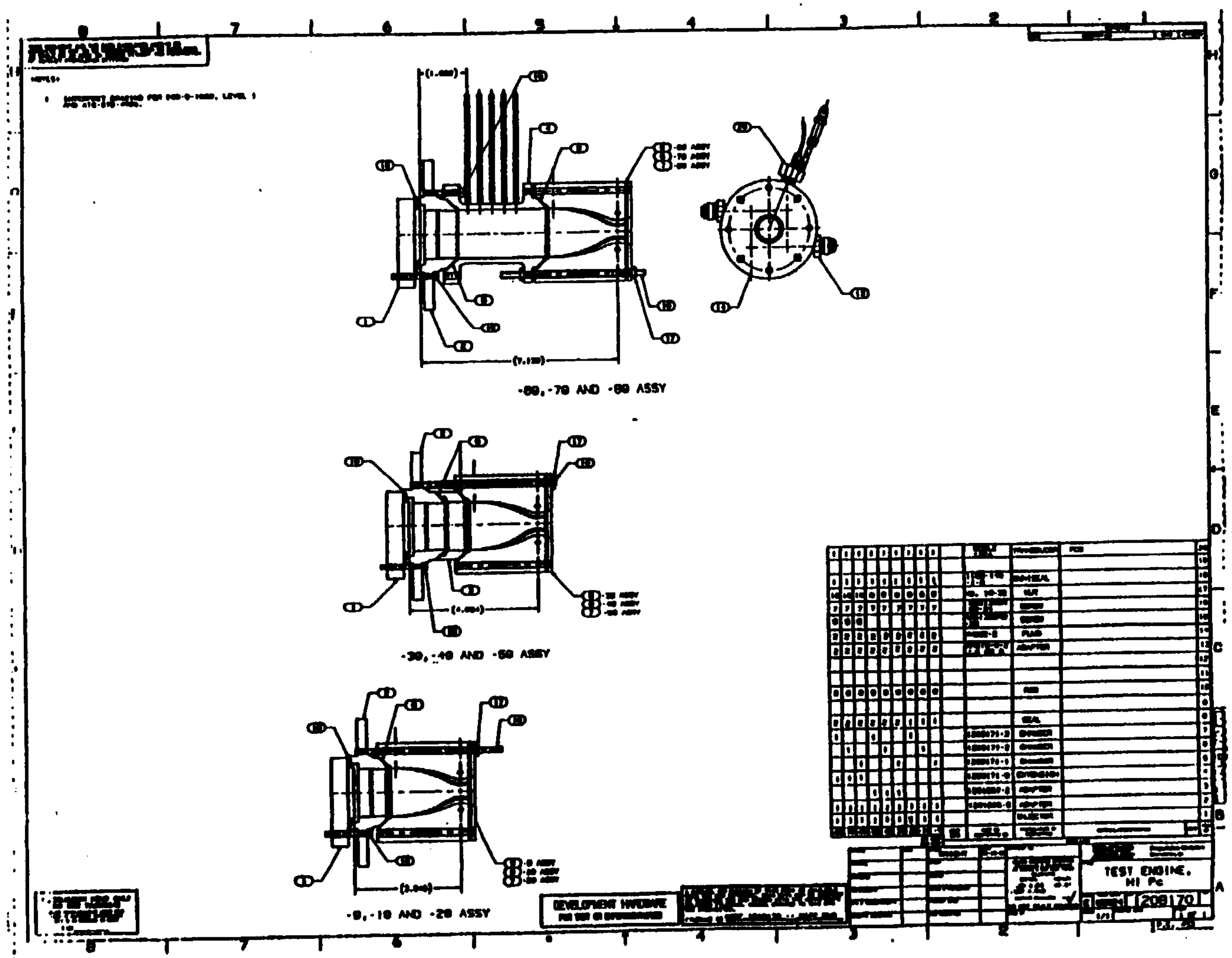
TEST GROUP	HARDWARE TRIP	COOLING	THROAT	CHAMB. EXT.	OPERATING CONDITIONS Pc	MR	DATA MEASUREMENTS PERFORM.	THERMAL	STABIL.	PLUME EMISS.
-101 to -113	regen	water	-1, -2, -3	--	100, 250, 500	0.94 TO 1.22	yes	--	yes	yes
-114 to -118	regen	fuel	-3	--	100	0.77 TO 1.03	yes	--	yes	yes
-119 to -127	FFC	FC	-2,-3	--	100, 250	0.92 TO 1.24	yes	yes	yes	--
-128 to -131	FFC	FC	-3	--	100	0.9 TO 1.27	yes	yes	yes	--
-132 to -142	FFC	FC	-1, -2, -3	--	100, 250, 500	0.73 TO 1.29	yes	yes	yes	--
-143 to -146	FFC	FC	-2	yes	250	0.84 TO 1.35	yes	yes	yes	--

High Pressure Earth Storable
Rocket Technology Program

REGENERATIVELY COOLED TESTBED

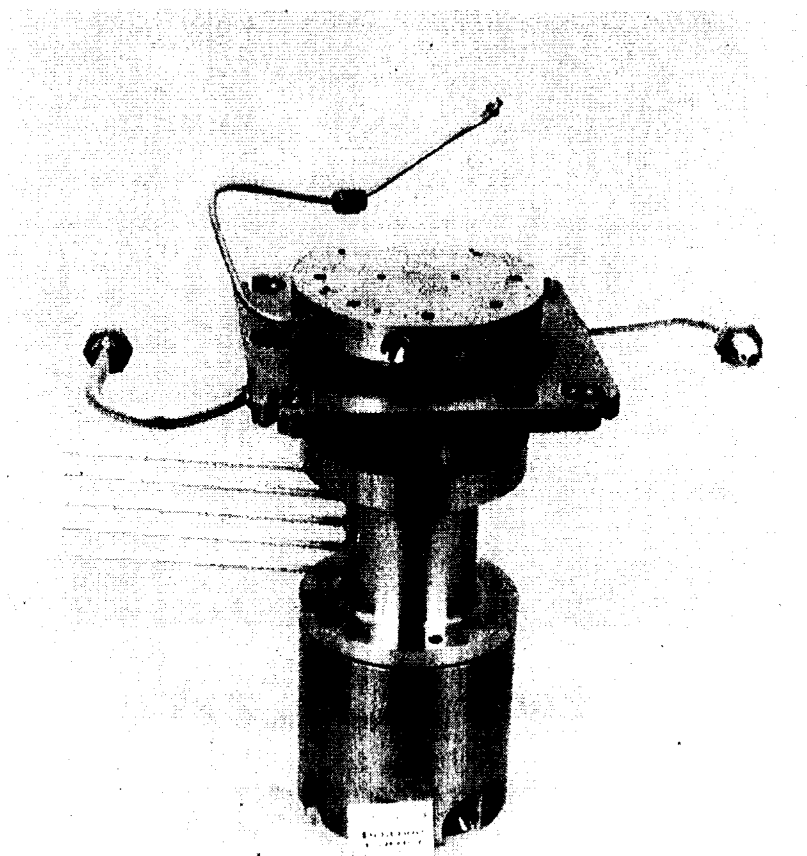
High Pressure Earth Storable Rocket Technology Program

Regen Cooled Testbed



High Pressure Earth Storable Rocket Technology Program

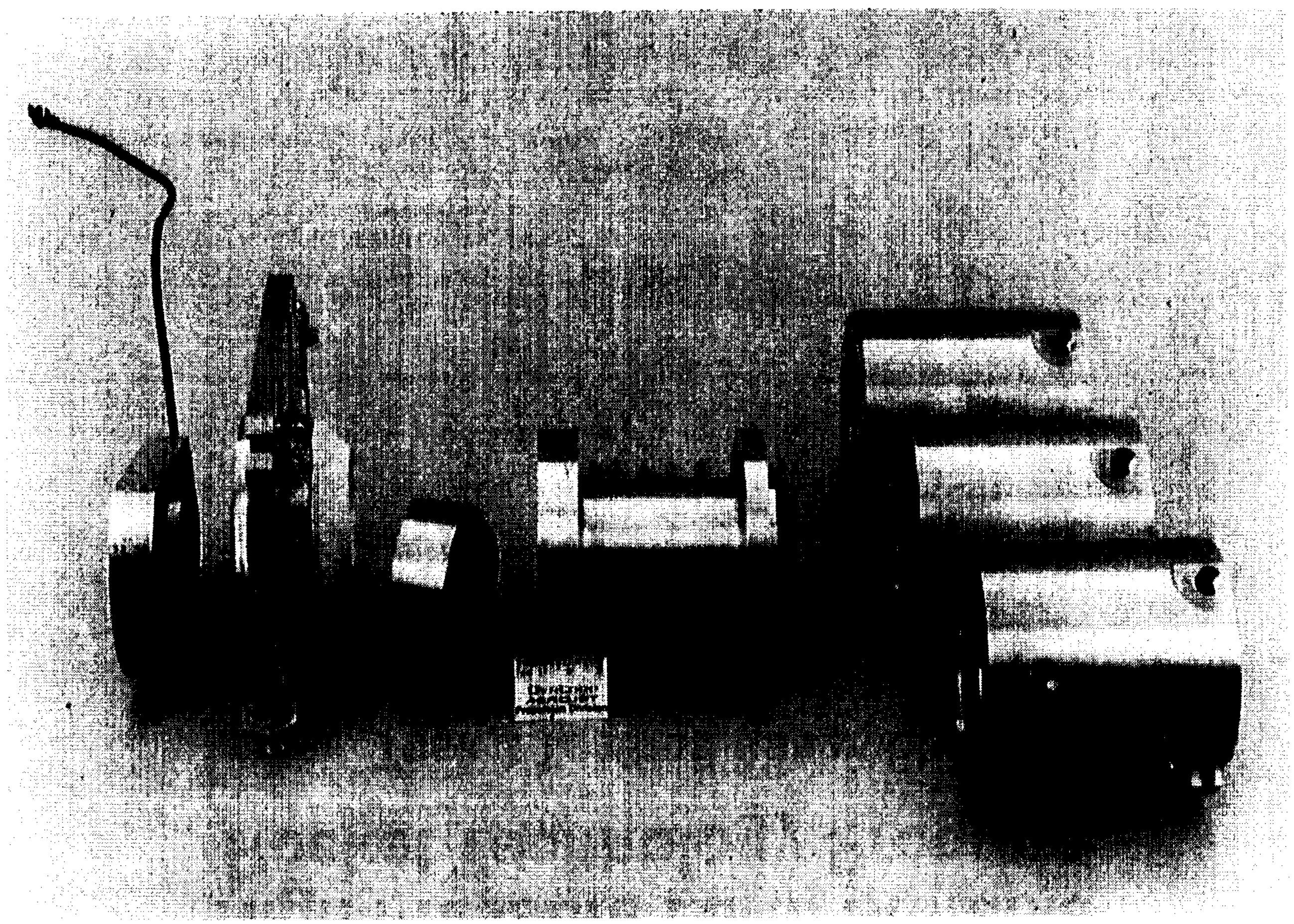
Task 2 Testbed Assembly



C12

High Pressure Earth Storable Rocket Technology Program

Task 2 Testbed Components



C12

High Pressure Earth Storable Rocket Technology Program

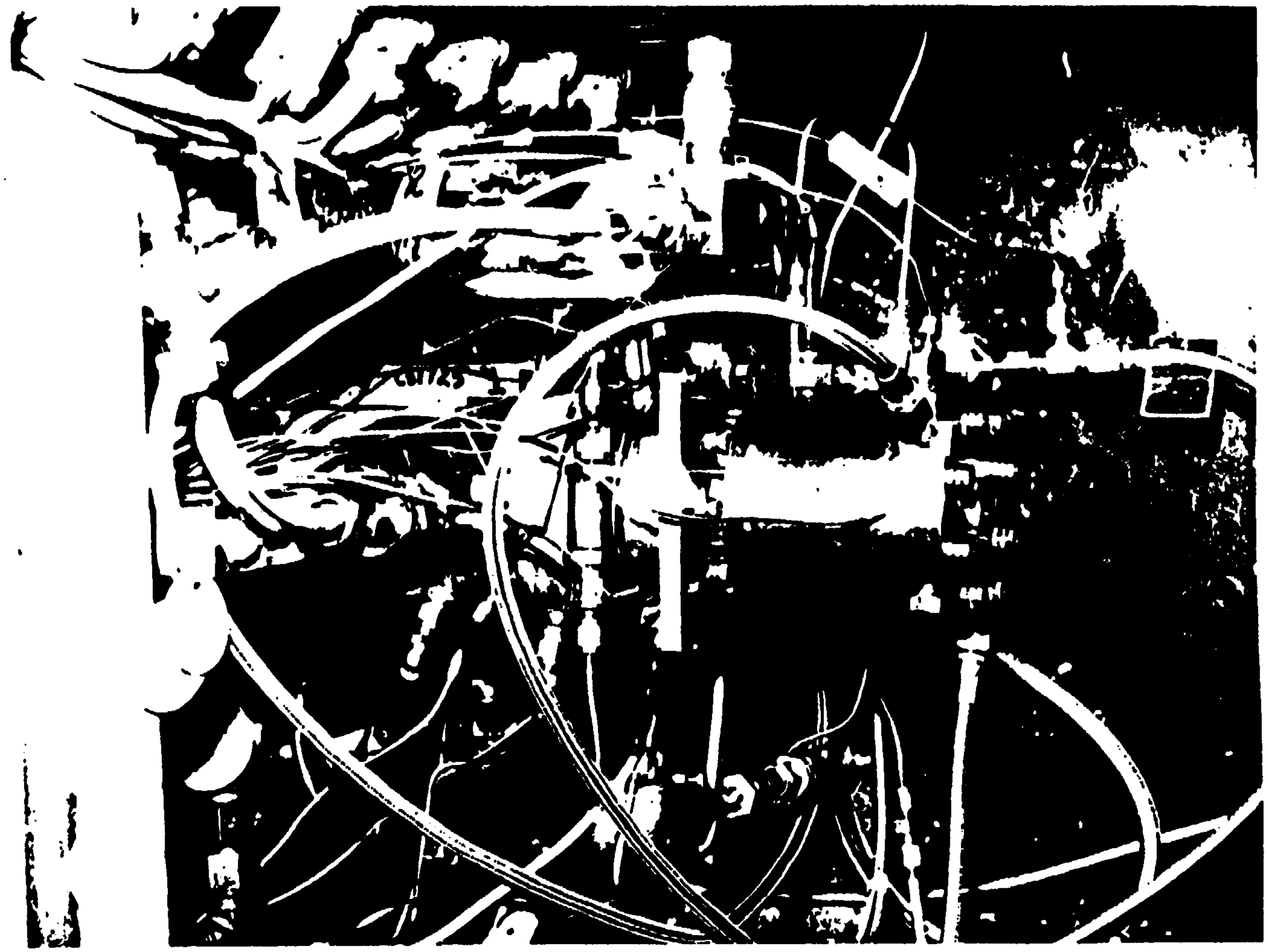
Injector S/N 6-1, 92 Element Platelet, with PCB Port

High Pressure Earth Storable Rocket Technology Program

Instrumented Chamber – Entrance View

High Pressure Earth Storable Rocket Technology Program

Task 2 Testbed Engine Setup in Bay A2



Regeneratively-Cooled Front End
Task 2 Test Results

17.130-4/5

High Pressure Earth Storable Rocket Technology Program

Task 2 Test Data Summary – Regen Cooled Front End

TEST	REGEN. COOLANT	THROAT AREA, IN^2	e, Ae/At	DATA TIME SEC	Pc-1 PSIA	MR O/F	Fsl LBF	Fvac LBF	Is vac SEC	C* FT/SEC	Cf vac --	Isp Pl 1.6:1	ERE	IspPl 300:1	Ispvac 300:1	COMMENT
101	WATER	0.525	1.60													CHECKOUT
102	WATER	0.525	1.60													CHECKOUT
103	WATER	0.525	1.60	9.75	100.0	0.972	58.66	70.73	239.0	5709	1.347	246.1	0.971	328.9	319.5	
104	WATER	0.525	1.60	9.75	99.2	1.039	57.89	69.97	238.5	5709	1.343	246.0	0.969	330.3	320.1	
105	WATER	0.525	1.60	9.74	104.0	1.115	61.26	73.34	238.7	5714	1.343	245.6	0.972	331.1	321.8	
106	WATER	0.525	1.60	9.76	100.4	1.249	58.58	70.66	235.7	5657	1.340	243.8	0.967	330.5	319.5	
107	WATER	0.214	1.65	9.49	242.6	0.944	65.83	70.76	240.9	5704	1.358	249.4	0.966	333.0	321.7	
108	WATER	0.214	1.65	9.50	249.9	1.056	67.56	72.50	241.3	5745	1.350	249.9	0.966	336.3	324.8	
109	WATER	0.214	1.65	9.50	247.9	1.215	66.80	71.74	241.0	5752	1.347	248.9	0.968	338.3	327.5	
110	WATER	0.106	1.67	3.00	492.9	0.928	68.70	71.17	239.2	5725	1.343	246.1	0.972	335.0	325.5	ERRONEOUS TC KILL
111	WATER	0.106	1.67	9.50	499.3	0.948	69.91	72.39	241.3	5754	1.349	250.6	0.963	335.8	323.4	
112	WATER	0.106	1.67	9.50	505.9	1.068	70.46	72.93	241.6	5793	1.341	251.5	0.961	339.9	326.6	
113	WATER	0.106	1.67	9.50	509.0	1.224	69.99	72.46	241.1	5856	1.324	250.9	0.961	342.7	329.4	
114	N2H4	0.525	1.60													CHECKOUT
115	N2H4	0.525	1.60	9.00	100.8	1.030	59.20	71.33	240.7	5747	1.347	246.1	0.978	330.1	322.9	
116	N2H4	0.525	1.60	4.78	95.4	0.773	55.08	67.22	233.8	5603	1.342	243.9	0.959	321.3	308.1	ERRONEOUS LOW Pc KILL
117	N2H4	0.525	1.60													KILL LOW Wo (BUBBLE)
118	N2H4	0.525	1.60	9.50	104.8	0.943	61.99	74.14	237.9	5682	1.347	246.0	0.967	328.1	317.3	POST TEST REGEN FAILURE

[1] CONFIG. A=MOOG SDI VALVE/S-N6-1 INJECTOR/COOLED ADAPTER/COPPER CHAMBER-THROAT

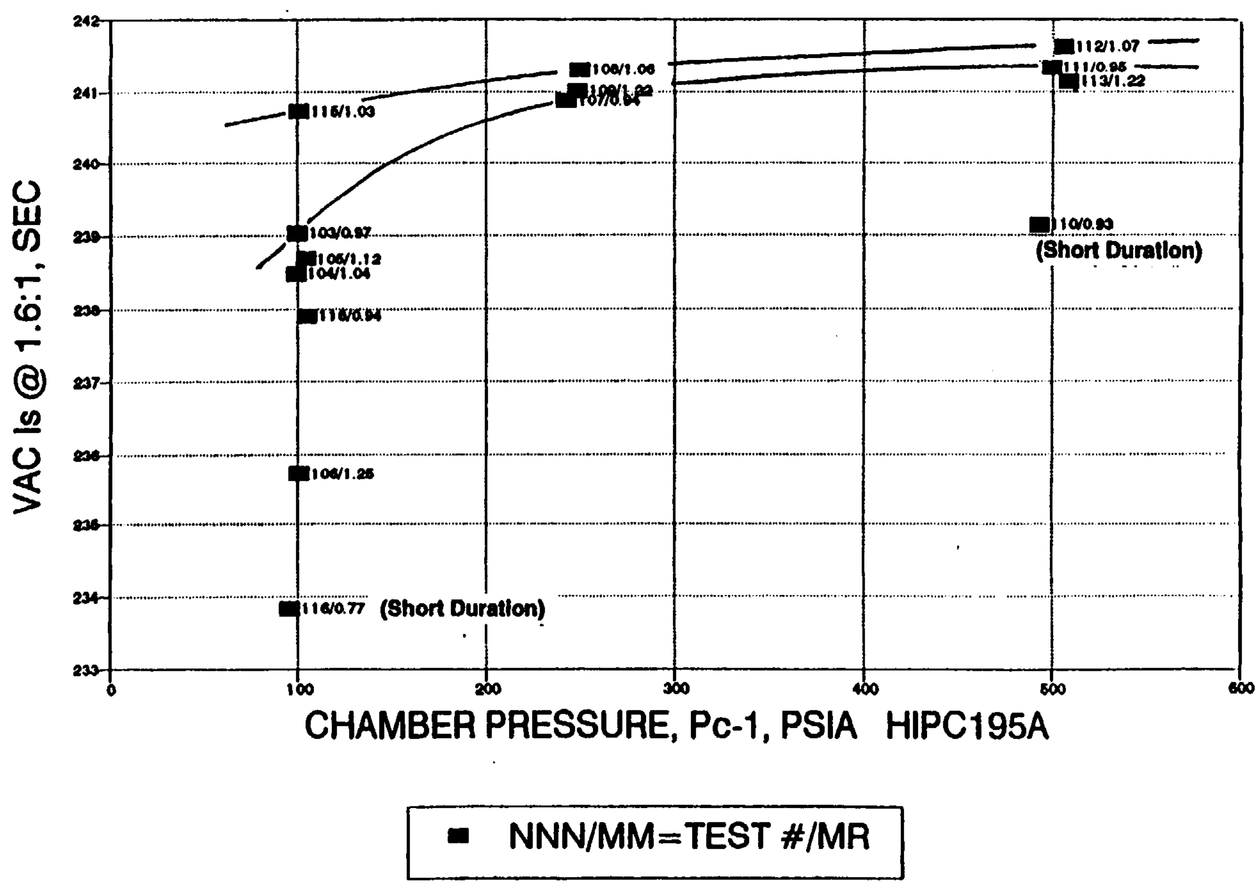
17.130-4/7

Vacuum Isp demonstrated with regeneratively cooled front end configuration and S/N 06-1 injector (thuster No. 1) at area ratio 1.6:1 increases significantly from Pc=100 to 250 psia (2-3 sec) but very little from 250 to 500 psia (<0.5 sec).

Note: Test Nos. 115, 116 and 118 were conducted with hydrazine flowing through the regen circuit; the other tests used water cooling.

High Pressure Earth Storable Rocket Technology Program

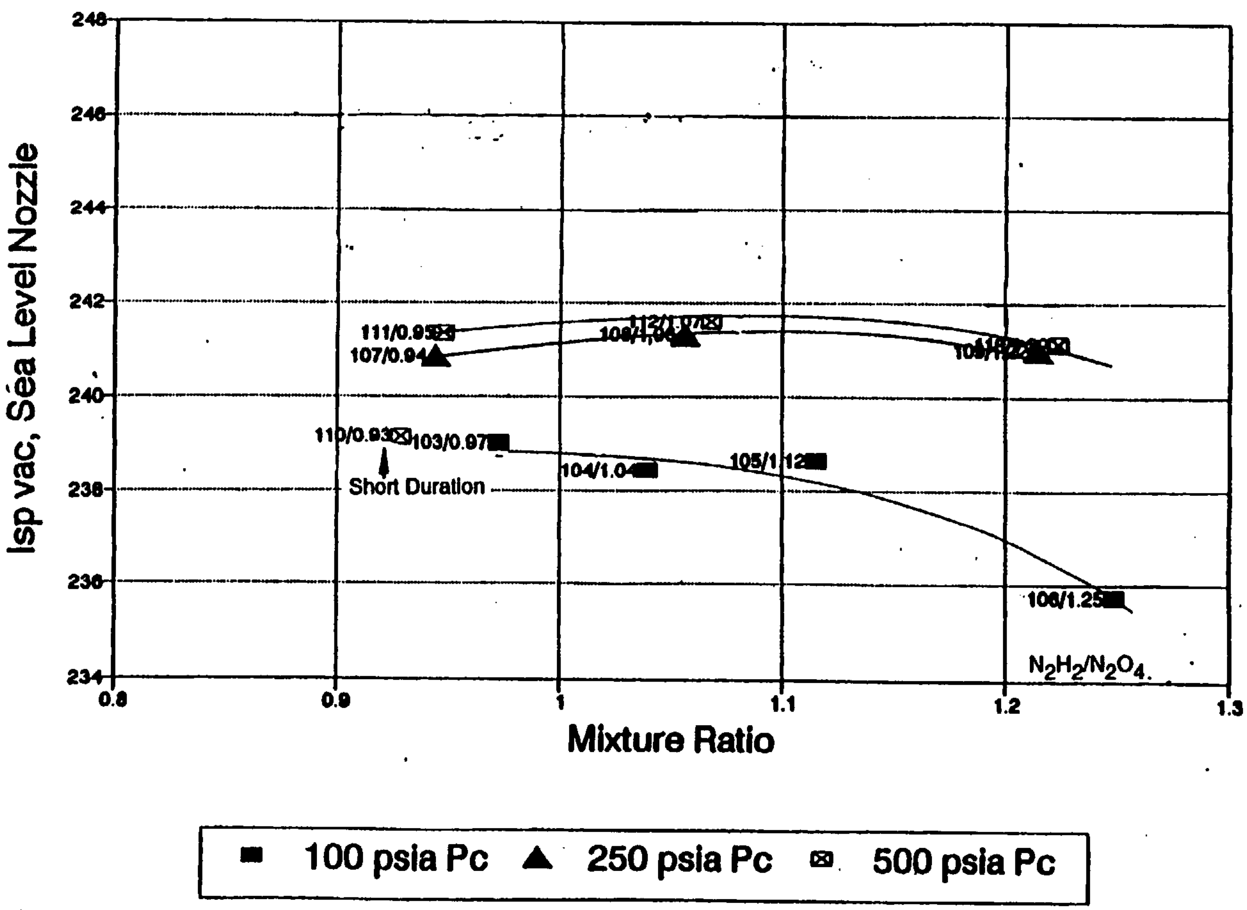
Effect of Chamber Pressure on Specific Impulse (For ε = 1.6:1 Nozzle at Vacuum)



Maximum vacuum Isp demonstrated with the regeneratively cooled front end configuration at area ratio 1.6:1 is at 0.9 to 1.1 depending on chamber pressure.

High Pressure Earth Storable Rocket Technology Program

Effect of Mixture Ratio on Measured Performance for ε = 1.6:1, Vacuum

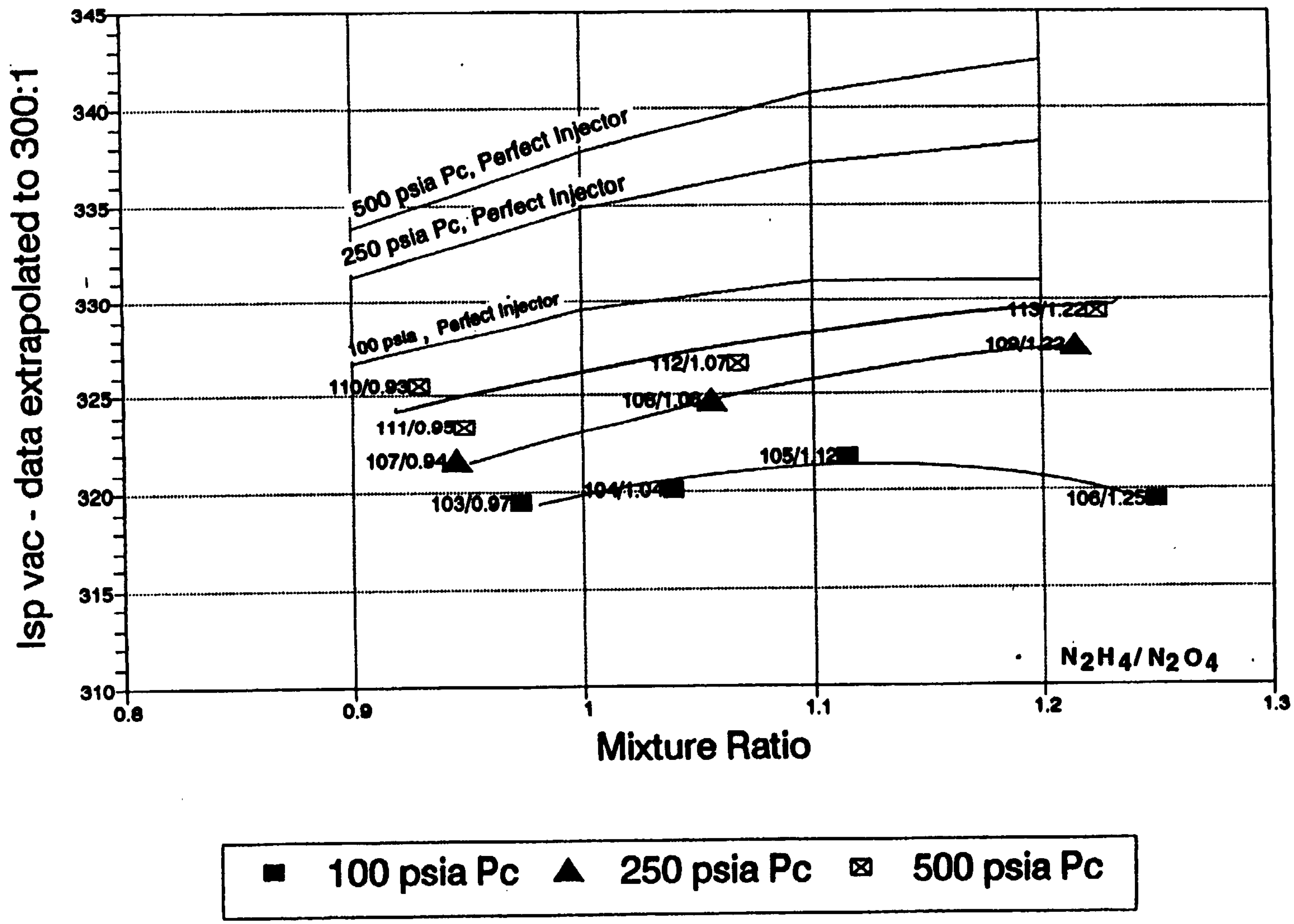


Predicted vacuum Isp at area ratio 300:1 with the regeneratively cooled front end configuration ranges from 321 to 329 sec depending on Pc level.

The optimum MR increased from the 0.9 to 1.1 range based on area ratio of 1.6:1 to 1.1-1.25 at 300:1.

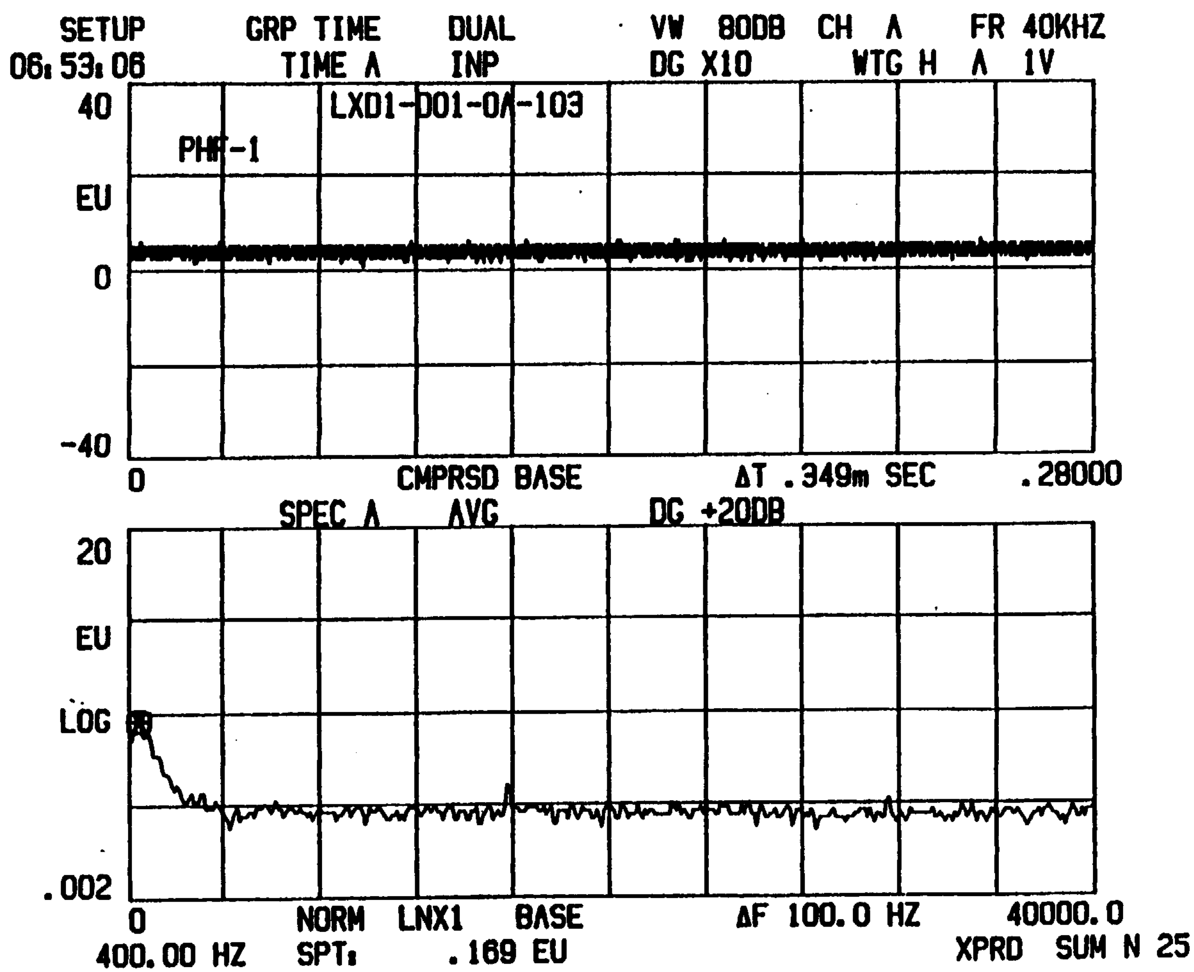
High Pressure Earth Storable Rocket Technology Program

Extrapolated Performance for 300:1 Nozzle



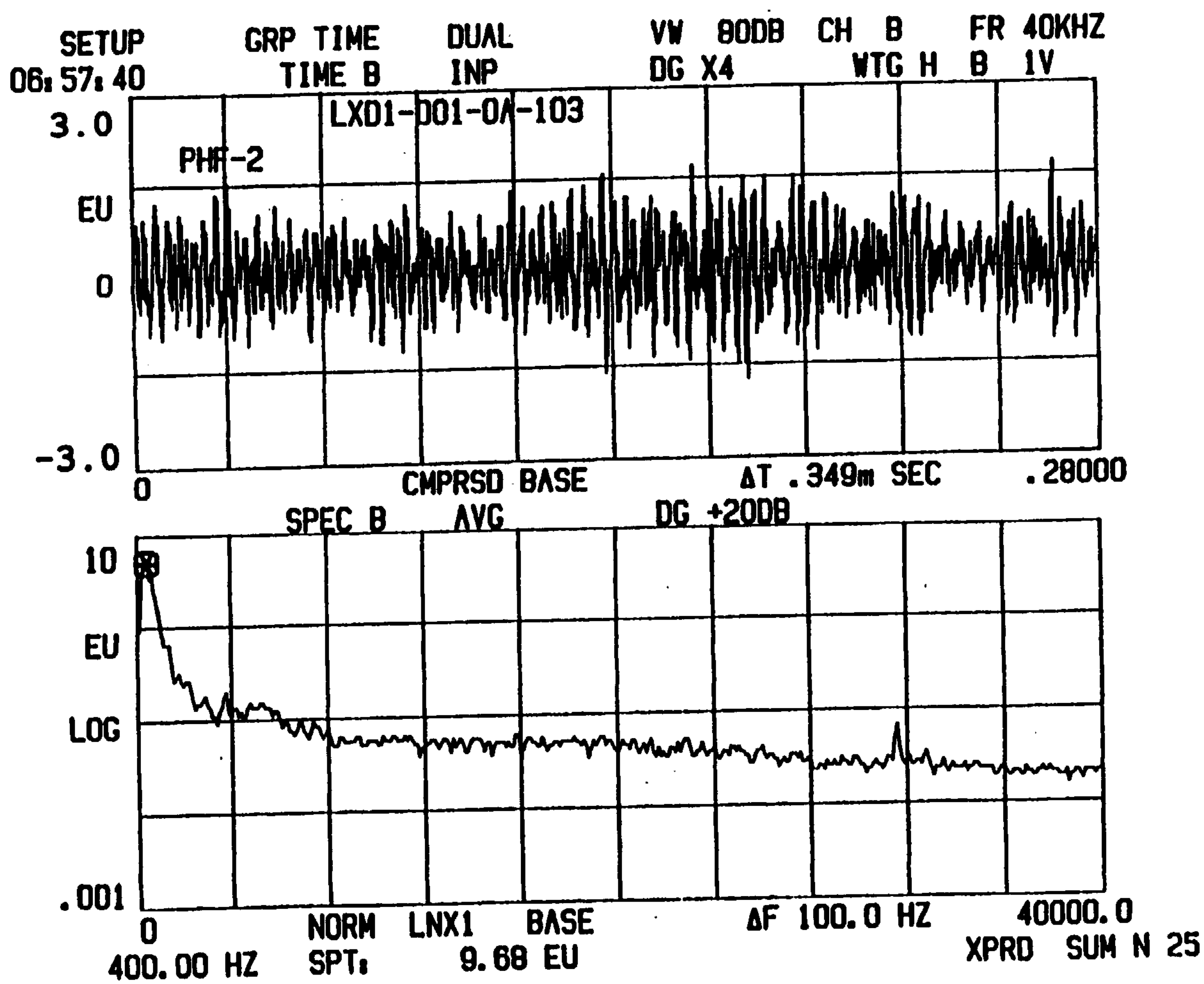
High Pressure Earth Storable Rocket Technology Program

Stability Data for Test -103, Injector Cavity



High Pressure Earth Storable Rocket Technology Program

Stability Data for Test -103, Chamber



High Pressure Earth Storable Rocket Technology Program

Plume Data Results Summary

- Distinct Spectra Were Recorded During Engine Firing; Before FS-1 and After FS-2 the Background Was (Essentially) Flat and Without Detail
- Spectra From OH and NH Were Expected. Spectra From CN, CH and C_2 Were Unexpected and Are Probably Due to a Propellant Line Contamination
- Intensities Varied Significantly with Pc and, to a Lesser Degree, Changes in MR
- Further Modeling Could Be Done to Predict Emission Intensities for Comparison to the Measurements

17.130-4b/rt/13

High Pressure Earth Storable Rocket Technology Program

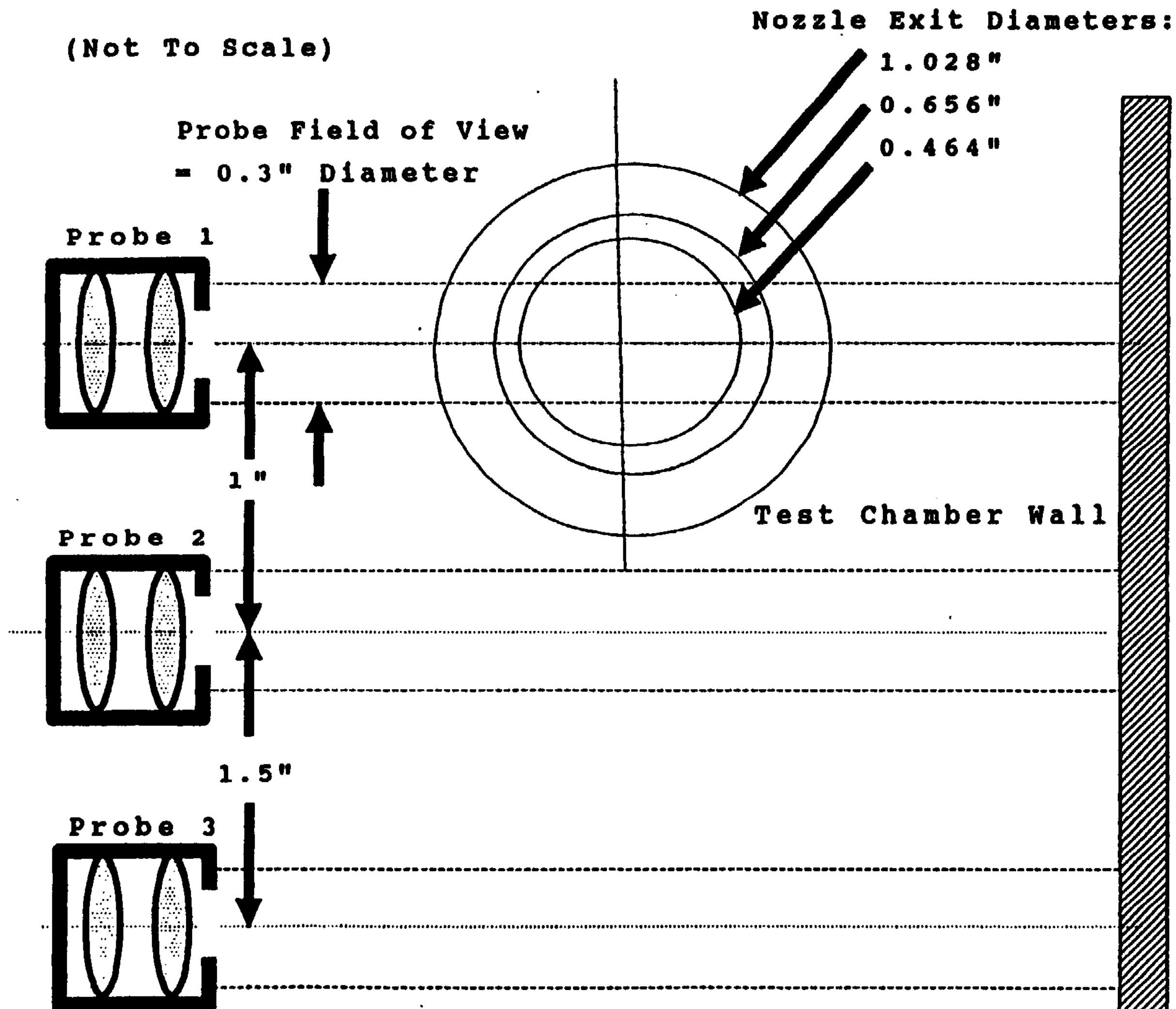
Plume Measurement Approach

- Three Fiber Optic Probes (0.3 in. Diameter Field of View)
- 0.25m Spectrometer Dispersed the Light
  (248 to 496 NM, 1 NM Resolution)
- Intensified CCD Camera Recorded the Spectra
  (0.1 sec Exposure Time)
- Hg Lamp Used for Wavelength Calibration

17.130-4b/rt/12

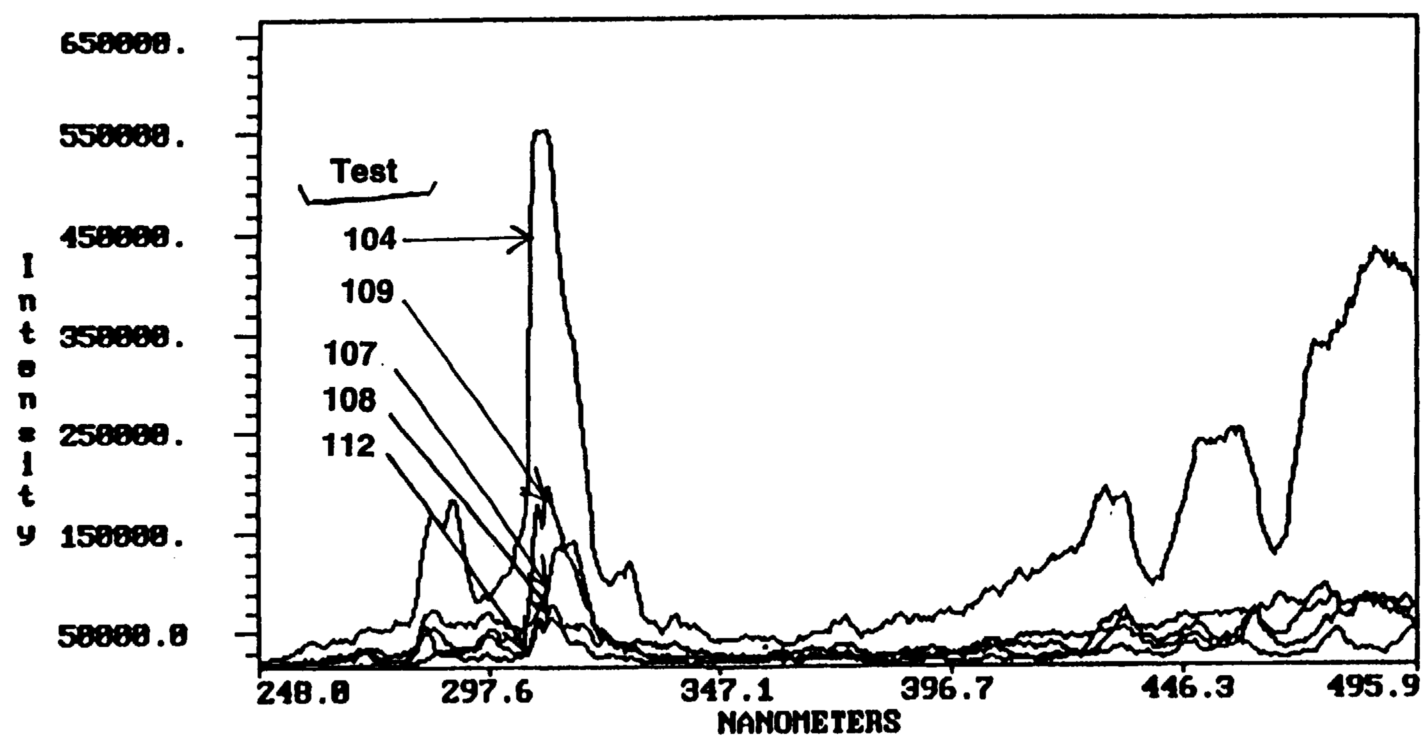
High Pressure Earth Storable Rocket Technology Program

Schematic of Optical Probe Alignment



High Pressure Earth Storable Rocket Technology Program

Composite of Spectra From Probe 1 Obtained During Engine Firing for Tests 104, 107, 108, 109 and 112



High Pressure Earth Storable Rocket Technology Program

Conclusions From Task 2 Regen Tests

1. Performance Improvement From 100 psia to 250 psia Pc Is Substantial; Increase, <u>for This Configuration</u>, is Not Significant at 500 psia Pc
2. Combustion Efficiency Is High
3. Combustion Is Stable
4. Thermal Management At Front End is a Problem With Fuel Regen Cooling

17.130-4/8

High Pressure Earth Storable Rocket Technology Program

Approach for Reliable Front End Thermal Management in Task 2 Exploratory Testbed Testing

- Explore Use of Fuel Film Cooled Trip/Front End
- Use FFC Hardware Built for NTO/Hydrazine Testbed IR&D Program
- Use Existing 92-Element Injector With Provision for Film Cooling (S/N 5)

17.130-4/9

High Pressure Earth Storable Rocket Technology Program

Task 2 Test Program

- Fuel-Film Cooled Front End
  - Hardware Configuration

17.130-4/10

- S/N 05 Injector Test Hardware Assembly

Following the post-fire detonation in the regenerative cooled chamber section (Test No. 118) the decision was made to continue the testing with a film cooled injector. Injector S/N 05 was available from a previous program and it also would interface the valve and chamber components selected for the test series. The adapter, ring, and trip were residual parts from an IR&D program and although all were made form CRES 300 series material they were deemed sufficient to withstand short duration tests (10 seconds maximum). The three copper chambers were designed for this test phase and had been used in Tests Nos. 101 through 118.

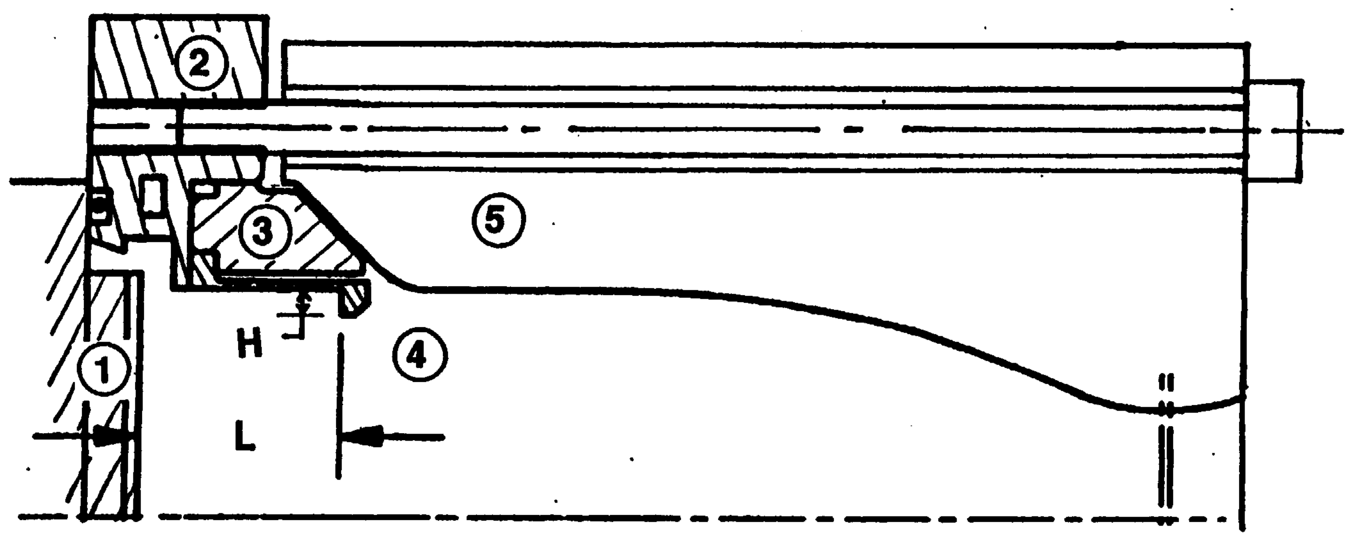
Trip and ring configurations were assembled and tested with the following lengths and heights:

- Combination No. 1: L = 1.00 in. and H = 0.10 in.
- Combination No. 2: L = 0.75 in. and H = .05 in.

A second test hardware configuration consisted of the basic assembly as shown but with a chamber extension, P/N 1208171-9, which increased the chamber length by 3.2 in. A Moog, Inc. valve, Model 53X186, was used for all tests.

High Pressure Earth Storable Rocket Technology Program

S/N 05 Injector Test Hardware Assembly

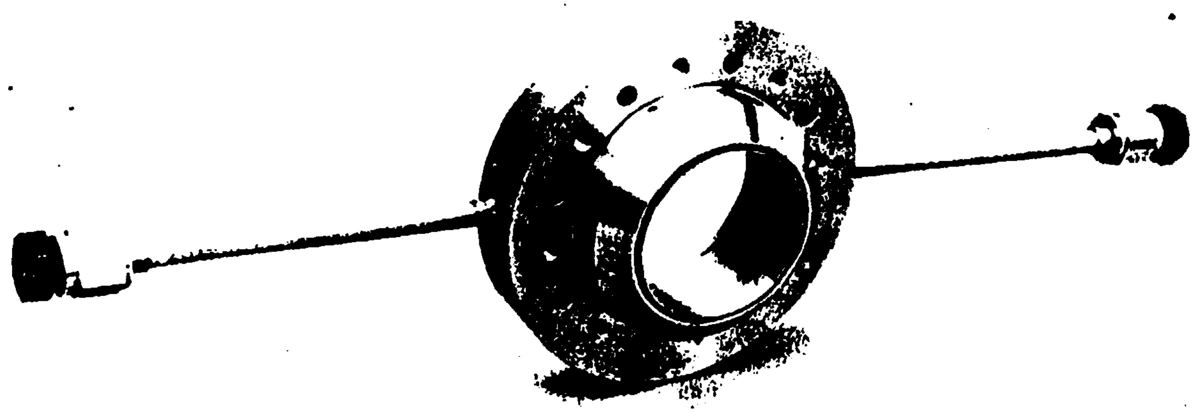


Item	P/N and Description
1 Injector	P/N 1206358-9, S/N 05: Platelet Design With Machined Manifold, CRES 300 Series
2 Adapter	P/N 1207296-9, Cooled Body, CRES 304
3 Ring	P/N 1207294-1 and -2, CRES 304
4 Trip	P/N 1207293-2 and -6, CRES 304
5 Chamber	P/N 1208172-1, -2, and -3, Copper

7.81.E8.1

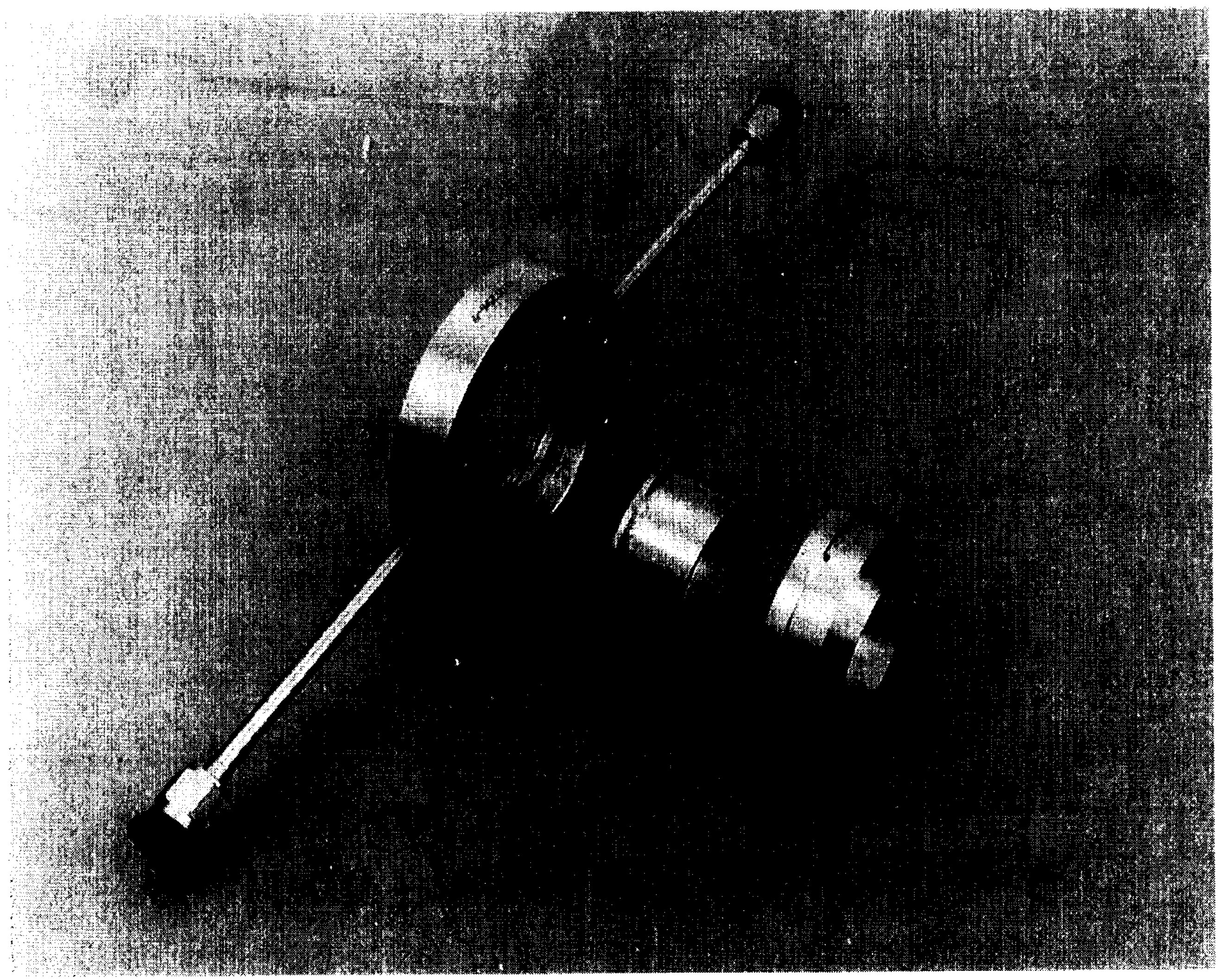
High Pressure Earth Storable Rocket Technology Program

Assembled Film Cooled Front End



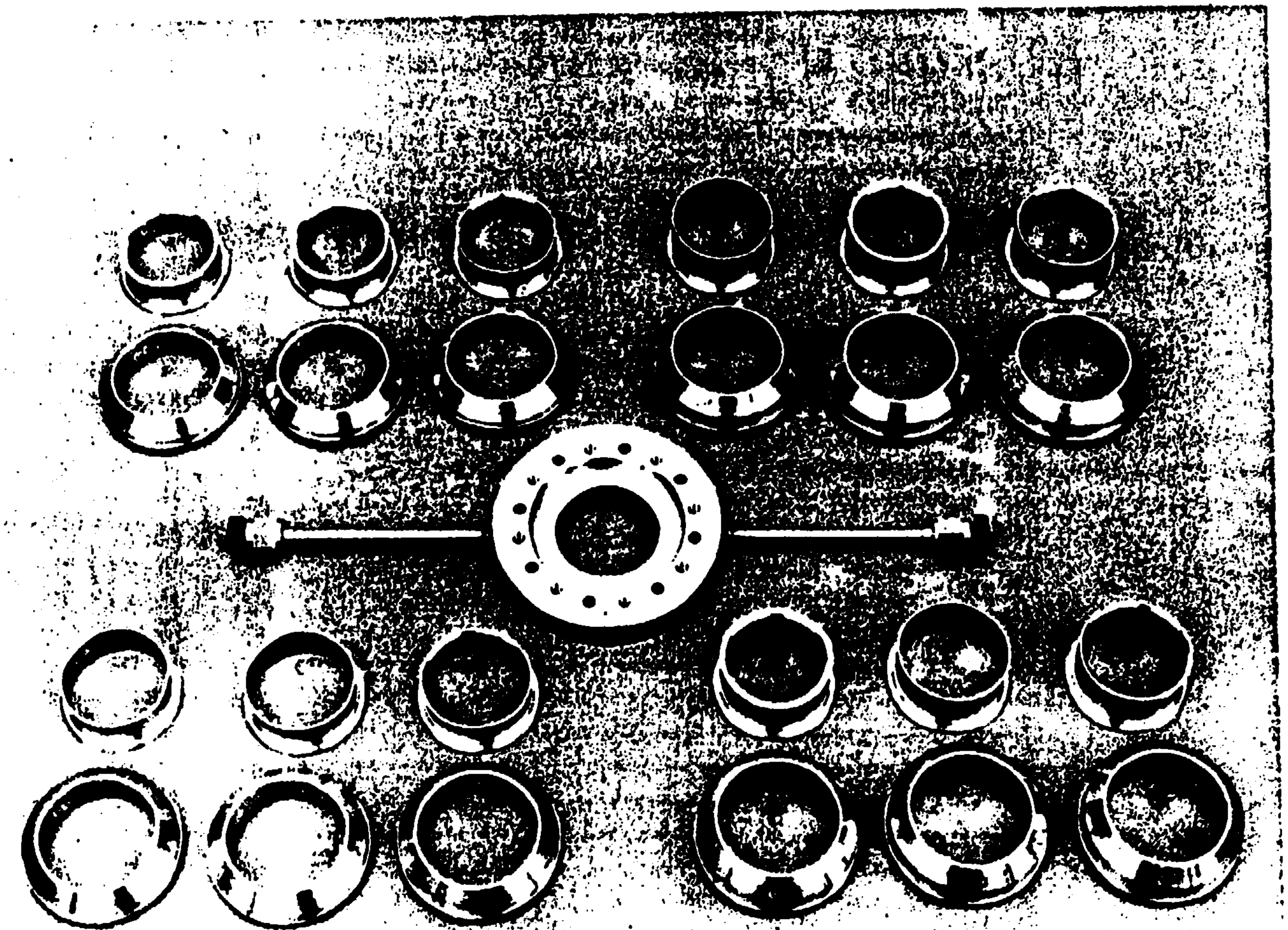
High Pressure Earth Storable Rocket Technology Program

Trip Section Prior to Installation of Thermocouples



High Pressure Earth Storable Rocket Technology Program

Hardware Available for Trip Ring Length and Height Survey



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- **Comparison of Critical Assembly Design Parameters**

Although the test hardware permitted testing over a large range of Pc's (100 to 500 psia) the thruster configurations were not at a common set of chamber parameters, i.e., contraction ratio (Ac/At) and characteristic length (L*). The values of these two parameters are near optimum for the Pc = 100 psia test point which is the design origin of the hardware. The test points at Pc = 250 and 500 psia had Ac/At and L* values that were much larger which results in the combustion gases traveling down the chamber at a lower velocity and resulting in a greater stay time. Both of these parameters have a significant effect on performance as well as the thermal characteristics. The higher Ac/At and lower velocities results in a more stable boundary layer which will not mix with the oxidizer rich core gases as well. The greater L* and stay time will allow for additional combustion to take place with higher Isp. The net effect on the performance comparison of the three Pc levels was quite small because the injector energy release efficiency is very high, ≈ 98%. The tests with the 3.2-in. chamber extension increased the L* from 25.6 to 59.5 in. (Pc = 250 psia) but the vacuum Isp only increased by one second.

17.130-4a/rt/8

High Pressure Earth Storable Rocket Technology Program

Comparison of Critical Assembly Design Parameters

Configuration	Nominal Pc	Ac/At	L'	L*	L*/L*100
1	100	4.3	3.2	10.5	1.0
2	250	10.5	3.2	25.6	2.4
3	500	21.1	3.2	51.7	4.9
4	250 With Chamber Extension	10.5	6.4	59.5	5.7

17.130-4/12

High Pressure Earth Storable Rocket Technology Program

S/N 05 Injector

- **Design:**
  - P/N 1206358-9
  - Configuration: 8 Platelet Stack Bonded to a Machined Manifold (P/N 1206357)
  - Face Pattern: 3 Concentric Rows of 92 Doublet Elements Plus 3 Fuel Elements at Center
  - Doublet Elements: Ox-on-Fuel Splashplate Elements
  - Outer Row Elements: 48 Each Fuel Elements Are Modified to Direct ~ 20% of Flow to the Chamber Wall and ~ 80% to the Adjacent Oxidizer Stream
  - Additional Fuel Film Cooling: 16 Each Orifices in Resonator Cavity (Removable Fittings)
- **Test History:** Tested on NASA Contract 3-25646 With Rhenium Chamber (% FFC = 0 From 16 Each Orifices) Performance Was About 1.5% Lower Than the Standard (S/N 6) Injector; Compatibility Was Not Measured
- **Task 2 Selection Criteria:** Available, Film Cooled Injector Design With High Performance That Would Match Other Hardware Interfaces

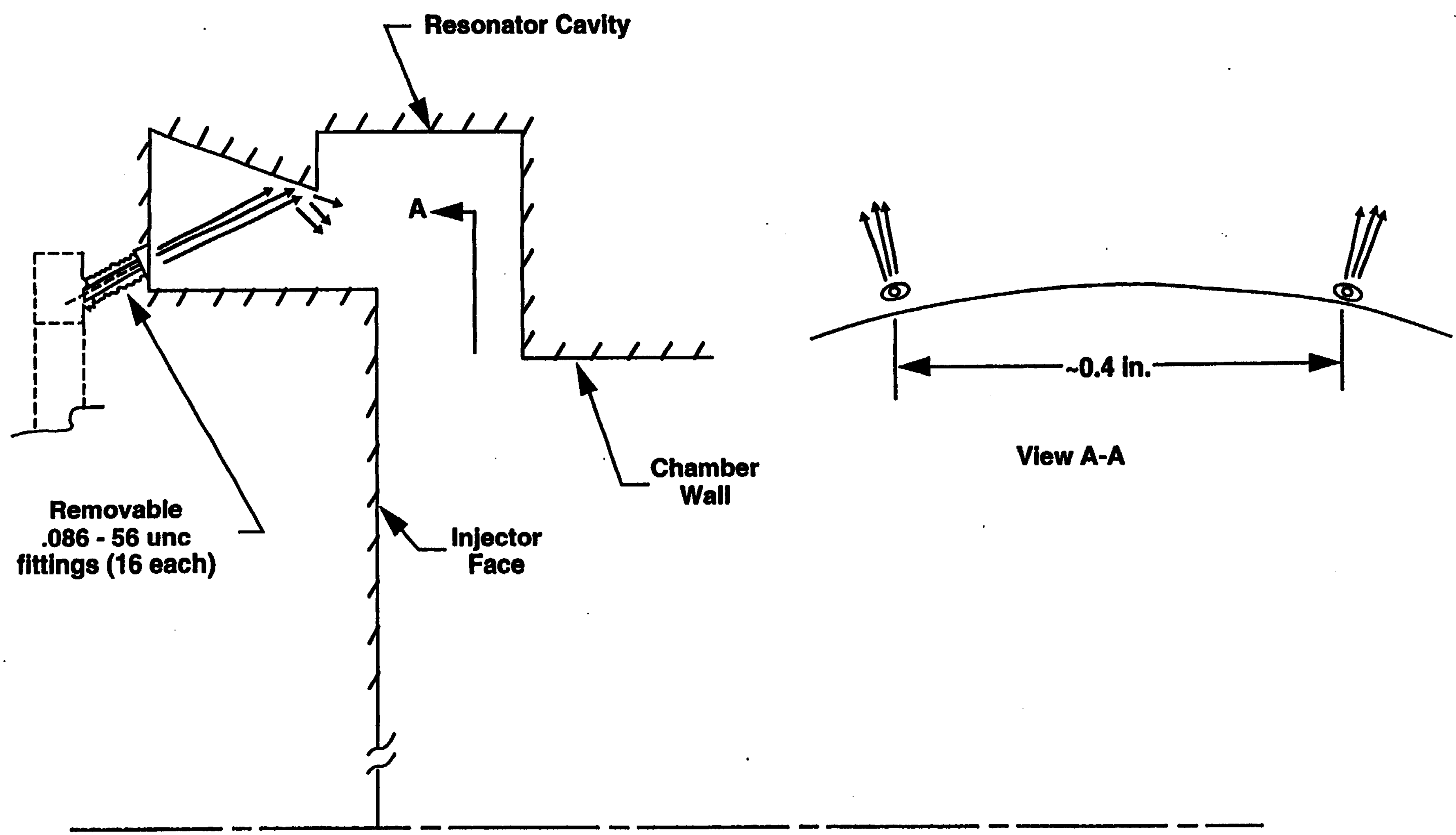
17.130-4/11

- **S/N 05 Injector Film Coolant Approach**

This basic injector concept was originally designed with a film coolant flow rate of ≈ 10% and is delivered from the outer row of fuel elements. On a previous 100-lbf test program, an increase in film coolant was required and was achieved with this basic injector design without redesigning the platelet stack by drilling 16, equally spaced, holes around the injector manifold. The film coolant is directed at a surface in the resonator cavity from where it flows onto the chamber wall. Consequently, the total fuel film coolant for those tests designated "42%" was actually about 50%. The 16 orifices are drilled into removable fittings which allows the film coolant flow rate to be easily changed.

High Pressure Earth Storable Rocket Technology Program

S/N 05 Injector Fuel Film Coolant Approach



7.81.ES.2

- **Task 2 Test Summary With Film Cooled Injector**

  Tests with Injector S/N 05 injector were conducted at several combinations of fuel film cooling percentages and trip configurations with the objective of reducing the thermal environment in order to achieve the target duration of 10 seconds without overheating the CRES 304 trip. The shortest length and smallest height trip configuration resulted in the most benign thermal conditions but durations of only 2-4 seconds at the Pc level of 500 psia were attainable before the trip kill temperature of 1500°F was achieved.

High Pressure Earth Storable Rocket Technology Program

Task 2 Test Summary With Film Cooled Injector

Date	Tests	Test Nos.	% FFC	Trip Configuration P/N	Trip Configuration Height, in.	Trip Configuration Length, in.	Comments
1/28/94	8	119-123 and 125-127	32%	1207293-6	0.10	0.92	Eroded small portion of trip on Test #121. Reduced kill temp to 1500°R Durations at Pc = 250 were ~2.0 sec. Did <u>not</u> test at Pc = 500
2/4/94	3	129-131	37%	1207293-6	0.10	0.92	No change in trip temps. All tests were at Pc = 100
2/11/94	10	132-135 and 137-142	42%	1207293-2	0.05	0.67	Conducted all tests as required. Test duration at Pc = 500 psia was 2.0-3.6 sec
2/14/94	4	143-146	42%	1207293-2	0.05	0.67	Conducted test with chamber extension at Pc = 250. Test duration on No. -146 was 8.4 sec as trip kill was set to 2200°F. Minor trip erosion

17.130-4/14

- **Test Series No. 1: 32% Fuel Film Cooling/0.10 in. Trip Height**

  Initial tests were conducted with 32% fuel film cooling and a trip length and height of 2.00 in. and 0.10 in. respectively. However, all but one of the 7 tests were less than 10 seconds. Minor trip erosion occurred with the trip kill temperature set at 1800°F, after which the kill limit was reduced to 1500°F.

High Pressure Earth Storable Rocket Technology Program

Test Series No. 1: 32% Fuel Film Cooling/0.10 Trip Height

						Quick Look Data*		
Test No.	Date	Copper Chamber	Firing Time (sec)	Pc (psia)	MR	Isp vac (sec)	C* (ft/sec)	Comments
119	1/28/94	1208172-3	1.0	102	0.92	---	---	Checkout Test
120	1/28/94	1208172-3	10.0	103	0.96	(237)	(5631)	Max Trip Test = 1535°F
121	1/28/94	1208172-3	8.1	99	1.11	(236)	(5620)	Computer shutdown on trip temperature (TTP-3A) @ 1800°F. Set kill at 1500°F. Observed eroded trip (over ~45° of trip arc)
122	1/28/94	1208172-3	5.5	99	1.24	(233)	(5580)	Computer shutdown on trip temperature (TTP-2B)
123	1/28/94	1208172-3	5.0	99	0.98	(236)	(5658)	Repeat of Test No. 119. Reduced duration to 5.0 sec
124	1/28/94	1208172-2	0.5	---	---	---	---	Inadvertent computer kill (Pc max limit too low)
125	1/28/94	1208172-2	3.2	247	0.92	(235)	(5604)	Computer kill on TTP-3A
126	1/28/94	1208172-2	2.2	250	1.08	(234)	(5638)	Computer kill on TTP-3A
127	1/28/94	1208172-2	1.8	246	1.23	(231)	(5578)	Computer kill on TTP-2B

*Quick look data is not corrected for final calibration adjustments and is therefore considered preliminary.

17.130-4/15

- **Test Series No. 2: 37% Fuel Film Cooling/0.10 Trip Height**

  Increasing the fuel film cooling from 32% to 37% did not significantly improve the test durations.

High Pressure Earth Storable Rocket Technology Program

Test Series No. 2: 37% Fuel Film Cooling/0.10 Trip Height

Test No.	Date	Copper Chamber	Firing Time (sec)	Pc (psia)	MR	Quick Look Data* Isp vac (sec)	C* (ft/sec)	Comments
128	2/4/94	1208172-3	0.5	---	---	---	---	Computer kill. No data from flowmeters (were disconnected)
129	2/4/94	1208172-3	10.0	101	0.90	237	5655	Max trip temp. at FS-2 (TTP-1A) = 397°F
130	2/4/94	1208172-3	5.8	100	1.09	235	5615	Computer kill on TTP-1A at 1500°F limit TTP-2A = 351°F TTP-3A = 343°F <u>Upstream Trip Temps</u>: TTP-1B = 218°F TTP-2B = 260°F TTP-3B = 220°F
131	2/4/94	1208172-3	4.5	98	1.27	231	5526	Computer kill on TTP-1A at 1500°F limit TTP-2A = 342°F TTP-3A = 730°F <u>Upstream Trip Temps</u>: TTP-1B = 238°F TTP-2B = 241°F TTP-3B = 236°F

*Quick look data is not corrected for final calibration adjustments and is therefore considered preliminary.

- **Test Series No. 3: 42% Fuel Film Cooling/.05 in. Trip Height**

  Test durations ranged from 2-10 seconds: the most severe trip thermal conditions were at the higher Pc and higher MR. It was observed that the computed Isp increased significantly from the 2.3 second summary period of the test to the 9-10 second summary ≈ 6 seconds. By comparison, the increase for the tests conducted with S/N 06 injector and the regenerative chamber section was ≈ 2 seconds. This difference is attributed to the fact that the film cooled chamber configuration has a much cooler boundary layer flow and as a result has a longer thermal transient.

High Pressure Earth Storable Rocket Technology Program

Test Series No. 3: 42% Fuel Film Cooling/.05 Trip Height

				Final Data[1]				
Test No.	Date	Copper Chamber	Firing Time (sec)	Pc (psia)	MR	Isp vac (sec)	C* (ft/sec)	Comments
132	2/11/94	1208172-3	10.0	97.4	.73	232.6	5600	@ 9-10 sec data slice: Ispv = 238.1
133	2/11/94	1208172-3	10.0	99.4	.92	233.3	5611	@ 9-10 sec data slice: Ispv = 238.6
134	2/11/94	1208172-3	10.0	98.8	1.02	233.0	5597	@ 9-10 sec data slice: Ispv = 237.4
135	2/11/94	1208172-3	7.7	99.0	1.29	227.9	5449	Computer kill on trip (TTP-1A) @ 1500°F; Ispv = 229.4 @ 7.25 - 7.75 sec
136	2/11/94	1208172-2	0.5	---	---	---	---	Computer kill. Incorrect Pc limits
137	2/11/94	1208172-2	5.0	248	.86	235.1	5584	Duration reduced to 5.0 sec; @ 4.5 - 5.0 Ispv = 236.4
138	2/11/94	1208172-2	5.0	247	1.08	232.9	5557	Duration reduced to 5.0 sec; @ 4.5 - 5.0 Ispv = 234.3
139	2/11/94	1208172-2	3.4	249	1.26	229.8	5523	Computer kill on TTP-3A[2]
140	2/11/94	1208172-1	3.6	497	.83	228.7	5489	Computer kill on TTP-3A[3]
141	2/11/94	1208172-1	2.5	492	1.05	224.9 @ 2.0-2.5 sec	5460	Computer kill on TTP-1A[3]
142	2/11/94	1208172-1	2.0	490	1.20	224.3 @ 2.0-2.5 sec	5410	Computer kill on TPP-1A[3]

Notes: (1) Final data has been corrected for final calibration adjustments. Data time slice is FS-1 + 2 to FS-1 + 3 seconds
(2) Two of three trip T/Cs exceeded 500°F
(3) All three trip T/Cs exceeded 500°F

- **Test Series No. 4: 42% Fuel Film Cooling/.05 Trip Height/Chamber Extension**

The chamber extension ΔL = 3.2 in., was only tested at Pc = 250 psia. The test durations on the initial three tests were only 3-3.6 seconds. Hence, the trip kill temperature was increased to 2200°F and the target MR reduced, 0.8-0.9. The subsequent test ran for 8.4 seconds which was sufficient to obtain meaningful thermal data.

High Pressure Earth Storable Rocket Technology Program

Test Series No. 4: Fuel Film Cooling/.05 Trip Height/Chamber Extension

Test No.	Date	Copper Chamber	Firing Time (sec)	Pc (psia)	MR	Final Data* Isp vac (sec)	Final Data* C* (ft/sec)	Comments
143	2/14/94	1208172-2	3.6	245	.98	237	5633	Computer kill on trip TTP-3A @ 1500°F (limit was increased to 1800°F)
144	2/14/94	1208172-2	3.5	247	1.05	236	5642	Computer kill on TTP-3A @ 1800°F (limit was increased to 2000°F)
145	2/14/94	1208172-2	3.1	247	1.35	231	5564	Computer kill on TTP-3A @ 2000°F (limit was increased to 2200°F)
146	2/14/94	1208172-2	8.4	246	.84	236	5697	Computer kill on trip TTP-3A @ 2200°F. Observed sparks in exhaust ~FS-1 + 5 sec
		Data @ FS-1 + 7.0 to 8.0 Sec		24			~~01	Trip erosion was insignificant

:onds

*Final data has been corrected for final calibrati

17.130-4/18

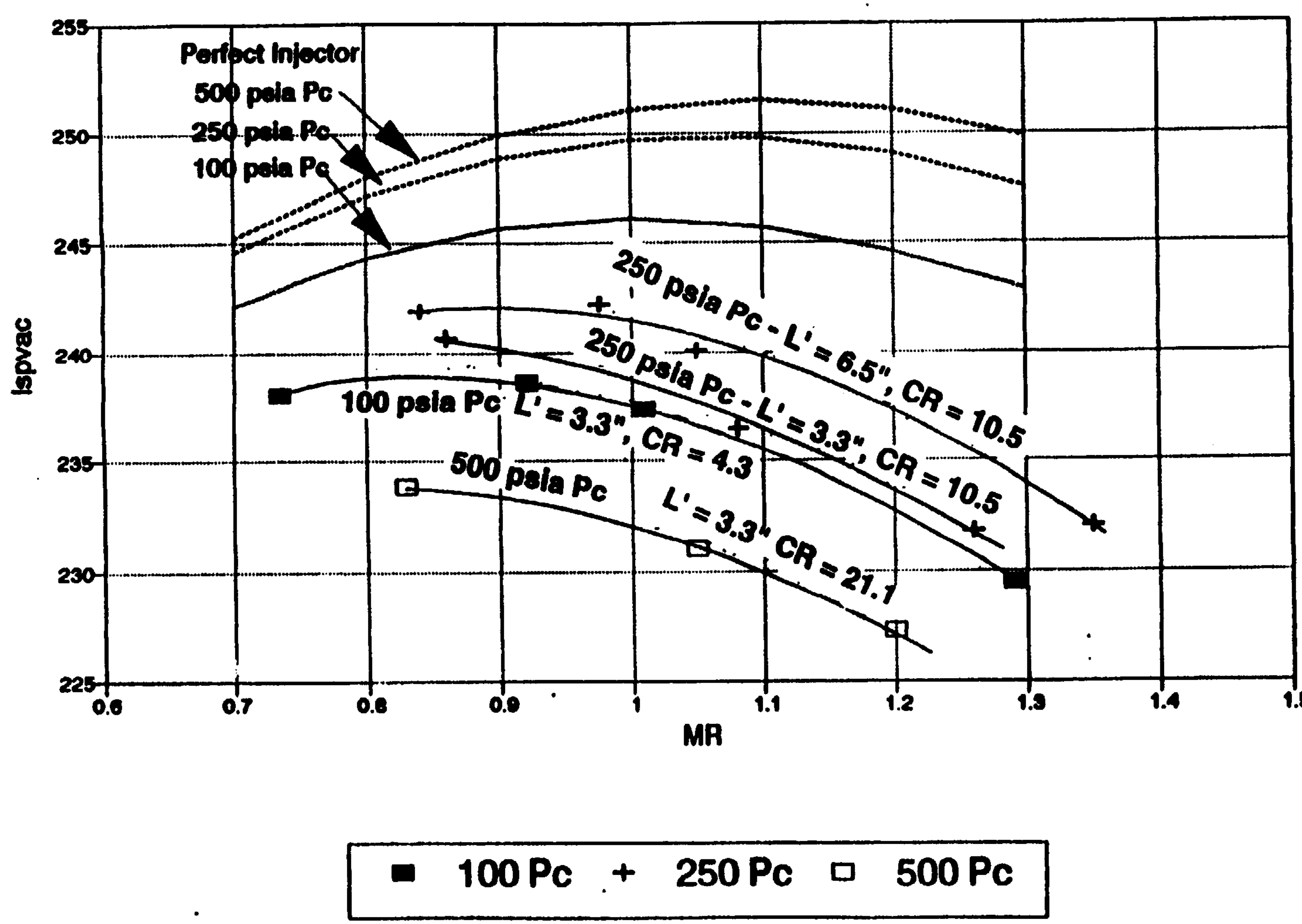
The maximum vacuum Isp at 1.6:1 area ratio demonstrated with the film cooled injector (S/N 5) and a chamber trip (Thruster No. 2) were the same as Thruster No. 1 at Pc levels 100 and 250 psia.

The vacuum Isp at Pc=500 psia was much less than for thruster No. 1 and, in fact, was less than the thruster No.2 Isp at Pc=100 psia. This is attributed to the loss in effectiveness of the trip to mix the fuel film coolant (42%) with the oxidizer-rich core gases. An optimum configuration should provide approximately the same results as Thruster No. 1.

The optimum MR was 0.8-0.9 which is lower than demonstrated with thruster No. 1 and is attributed to the fact that the fuel film cooling is not being mixed with the core gases.

High Pressure Earth Storable Rocket Technology Program

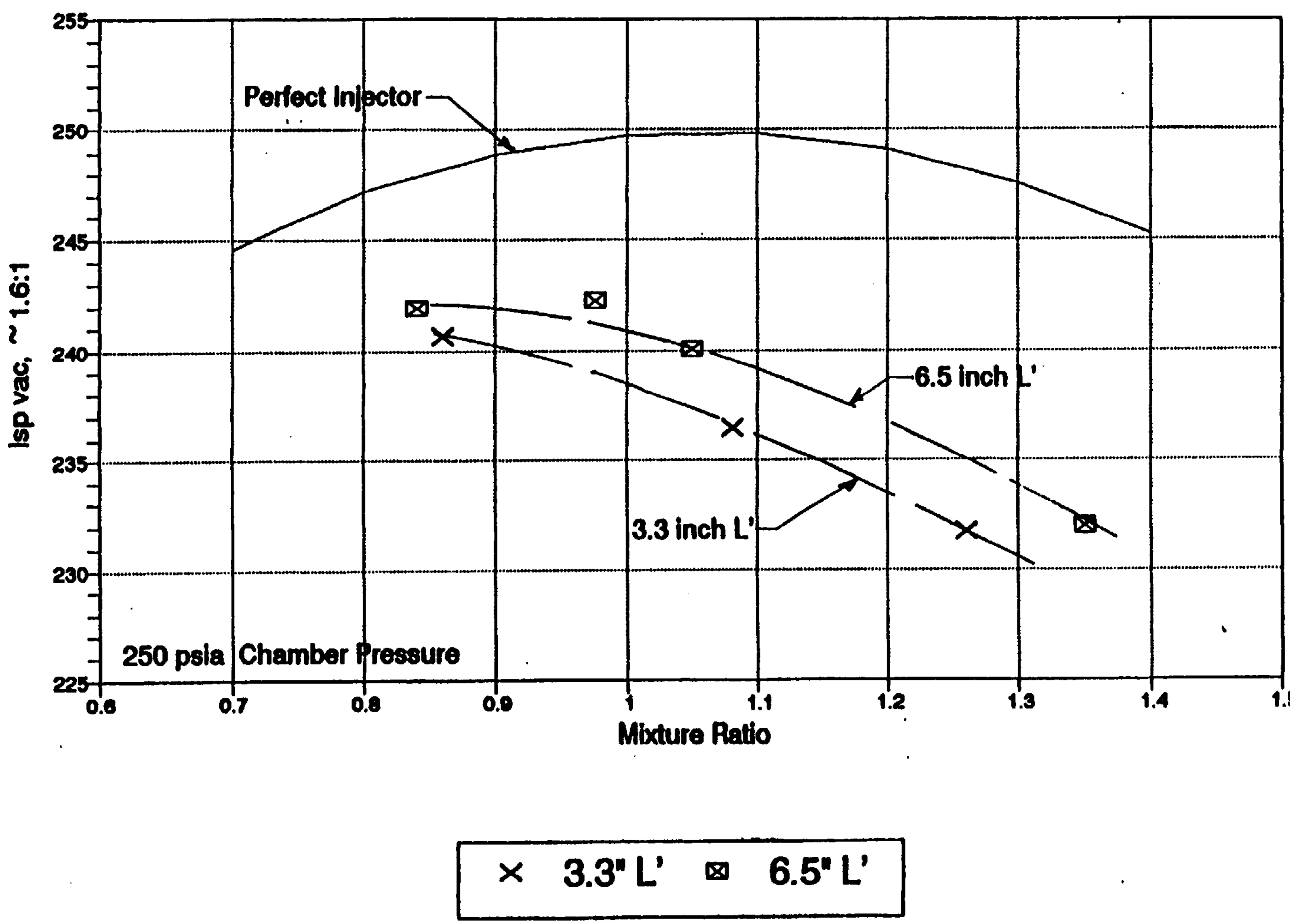
Vacuum 1.6:1 Isp for the 42% FCC Testbed



Thruster No. 2 tests at Pc=250 psia with chamber lengths (L') of 3.3 and 6.5 in demonstrated that vacuum Isp at 1.6:1 increased only about 1.0 sec with a doubling of length. This indicates that very little mixing and combustion takes place beyond an L' of 3.3 in.

High Pressure Earth Storable Rocket Technology Program

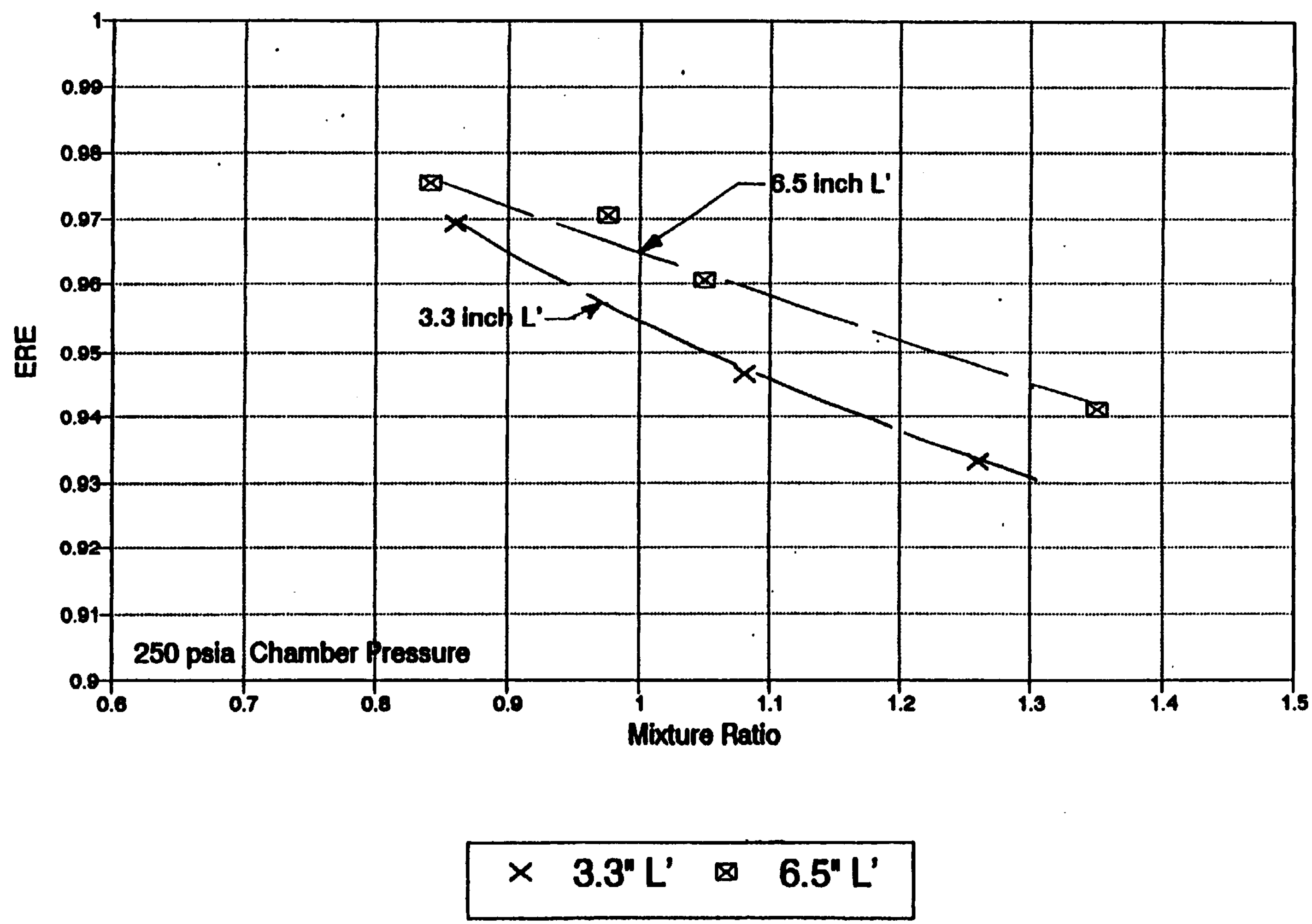
100 Lbf Data, 250 psia Pc
3.3" and 6.5" Chamber Length Data



The reduction in Thruster No. 2 energy release efficiency (ERE) at higher mixture ratios is further indication that the trip configuration is not optimum and does not mix all of the 42% fuel film coolant with the oxidizer-rich core (core MR=1.7 at an overall MR=1.0).

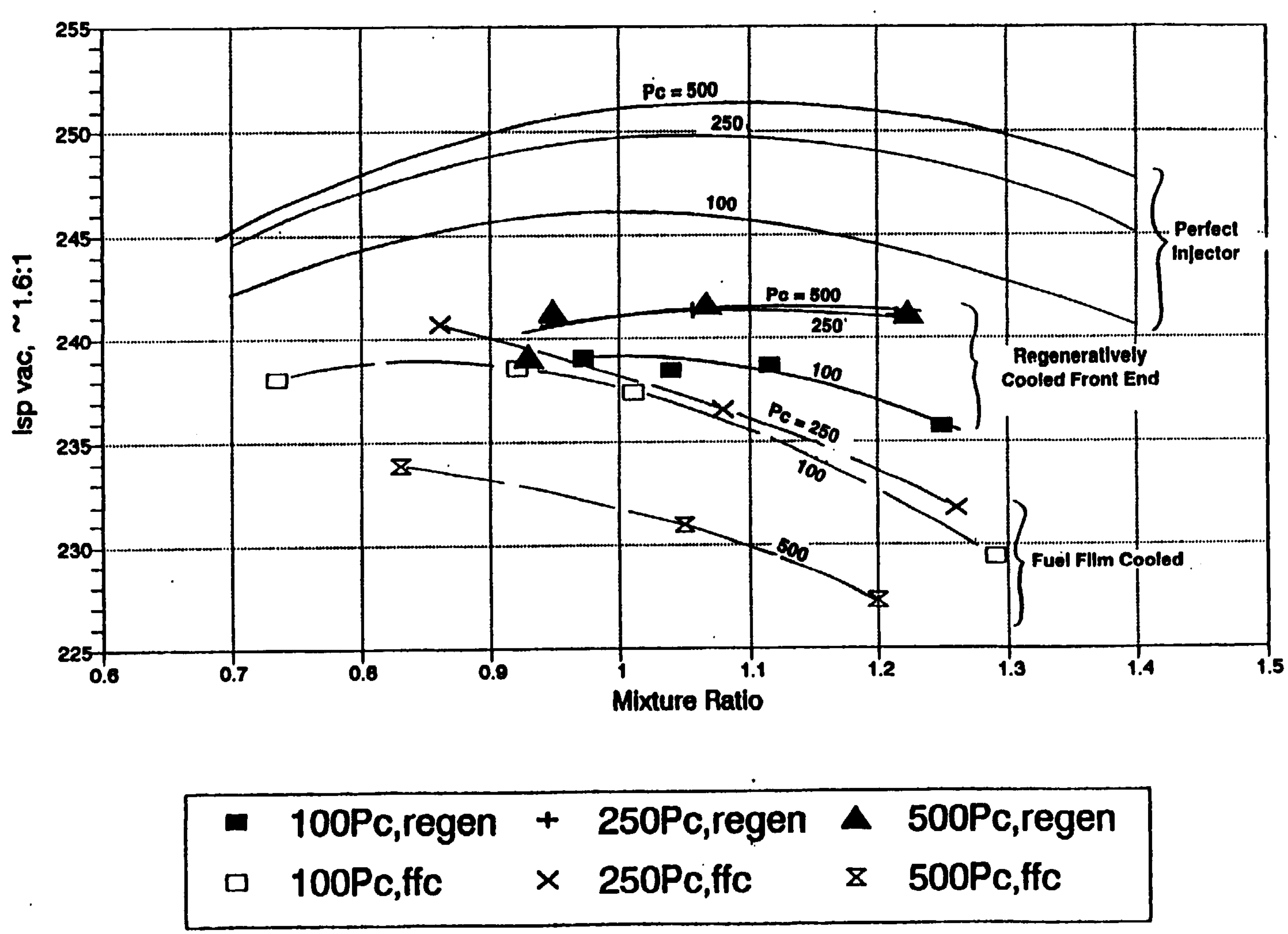
High Pressure Earth Storable Rocket Technology Program

100 Lbf Data, 250 psia Pc
3.3" and 6.5" Chamber Length Data



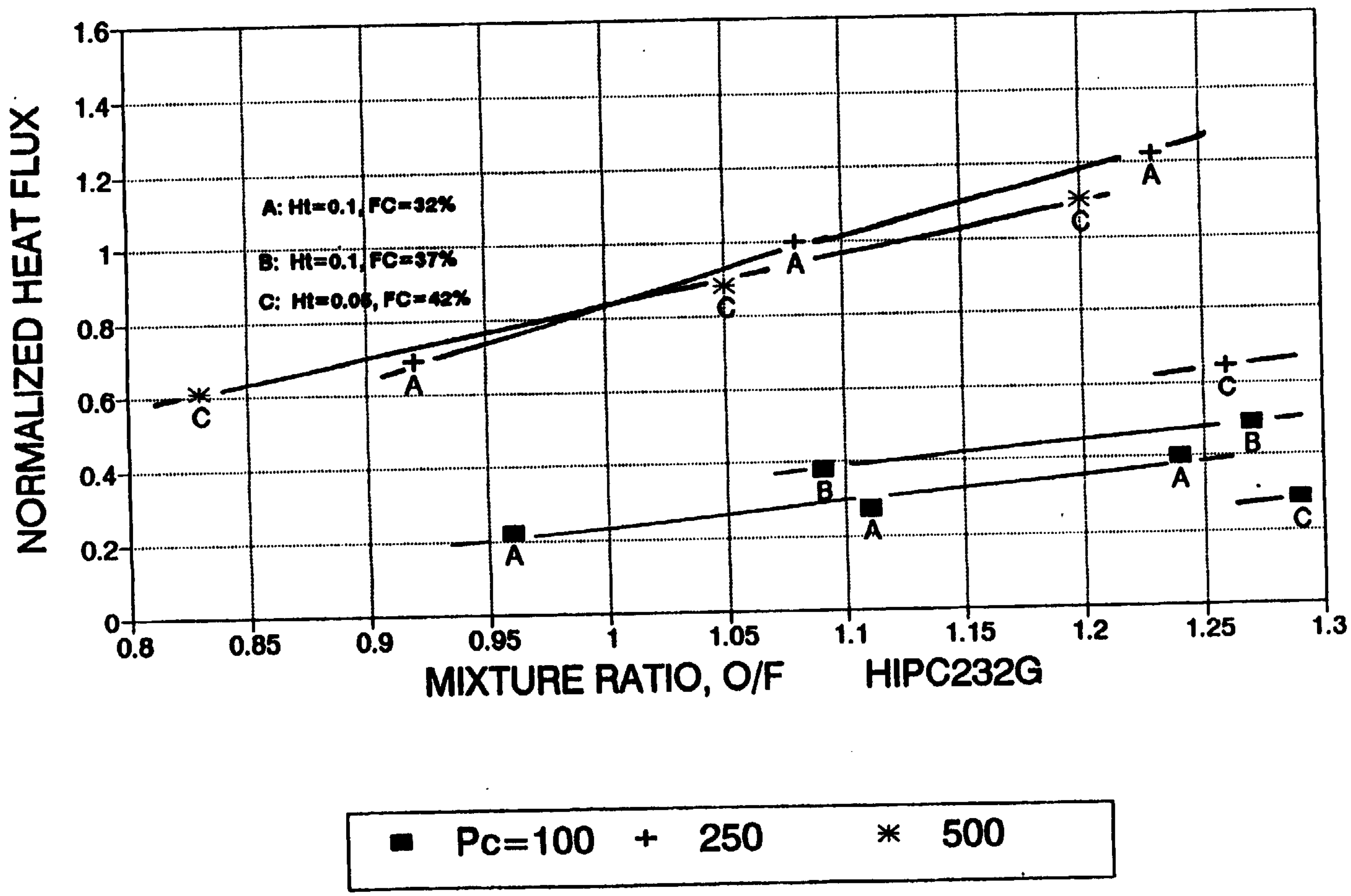
High Pressure Earth Storable Rocket Technology Program

100 Lbf Data, Pc from 100 psia to 500 psia Regen Chamber Data and 42% FFC Data



High Pressure Earth Storable Rocket Technology Program

NORMALIZED TRIP HEAT FLUX
TESTS -119 to -142 WITH 1500F KILL



High Pressure Earth Storable Rocket Technology Program

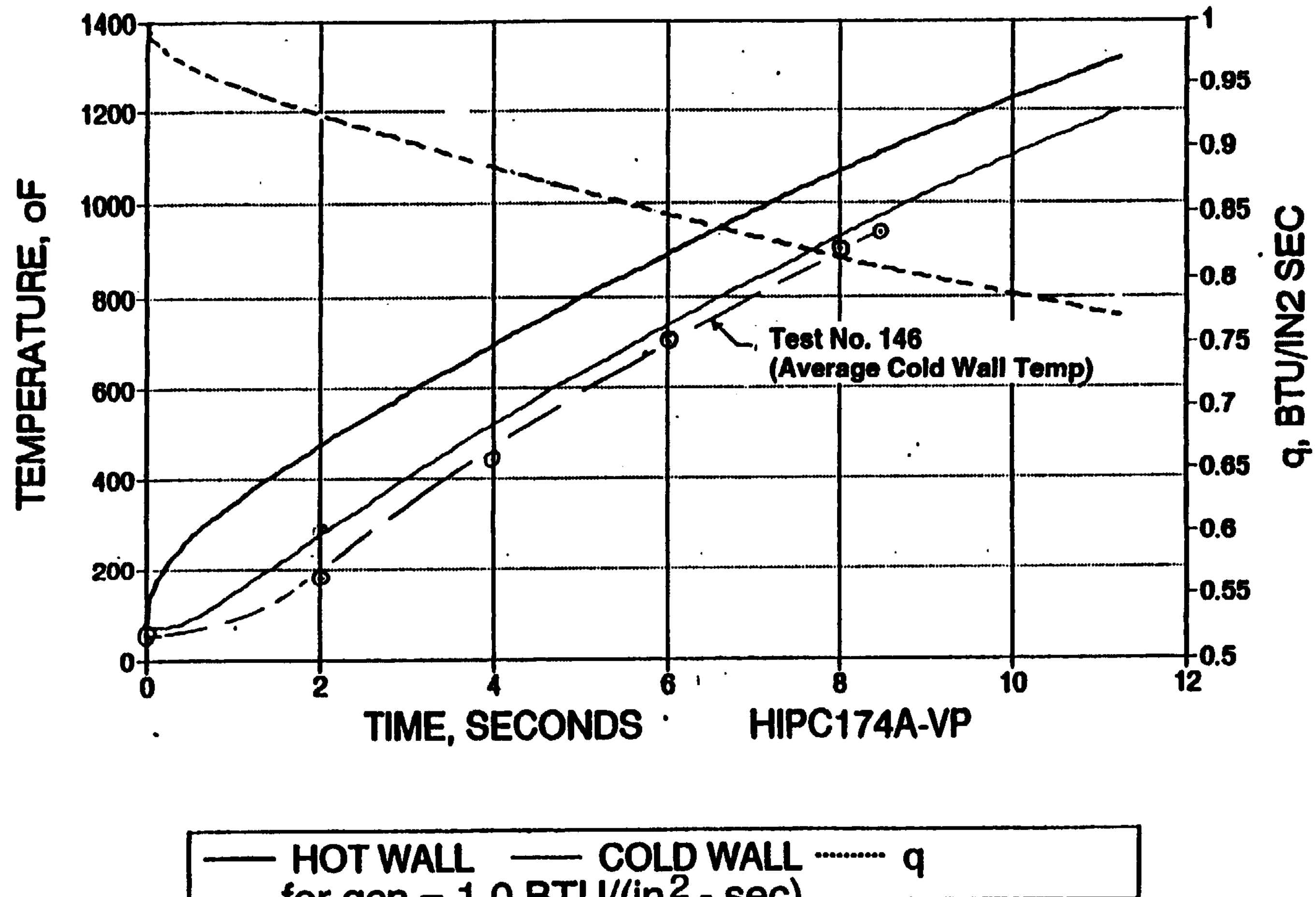
Trip Thermal Management Conclusions

- Generally Trip Can Be Maintained at Suitable Temperature (350°F)
- Locally S/N 5 Injector Gave Oxidizer Impingement Ahead of Trip, Limiting Test Durations With Stainless Steel Trip
- Acceptable Trip Height, Length and Percent Film Cooling Have Been Determined
- Optimum Values Not Yet Determined
- Potential Issue: Interaction Between Percent F.C. and Extent of Oxidizer Penetration to Wall

17.130-4/19

High Pressure Earth Storable Rocket Technology Program

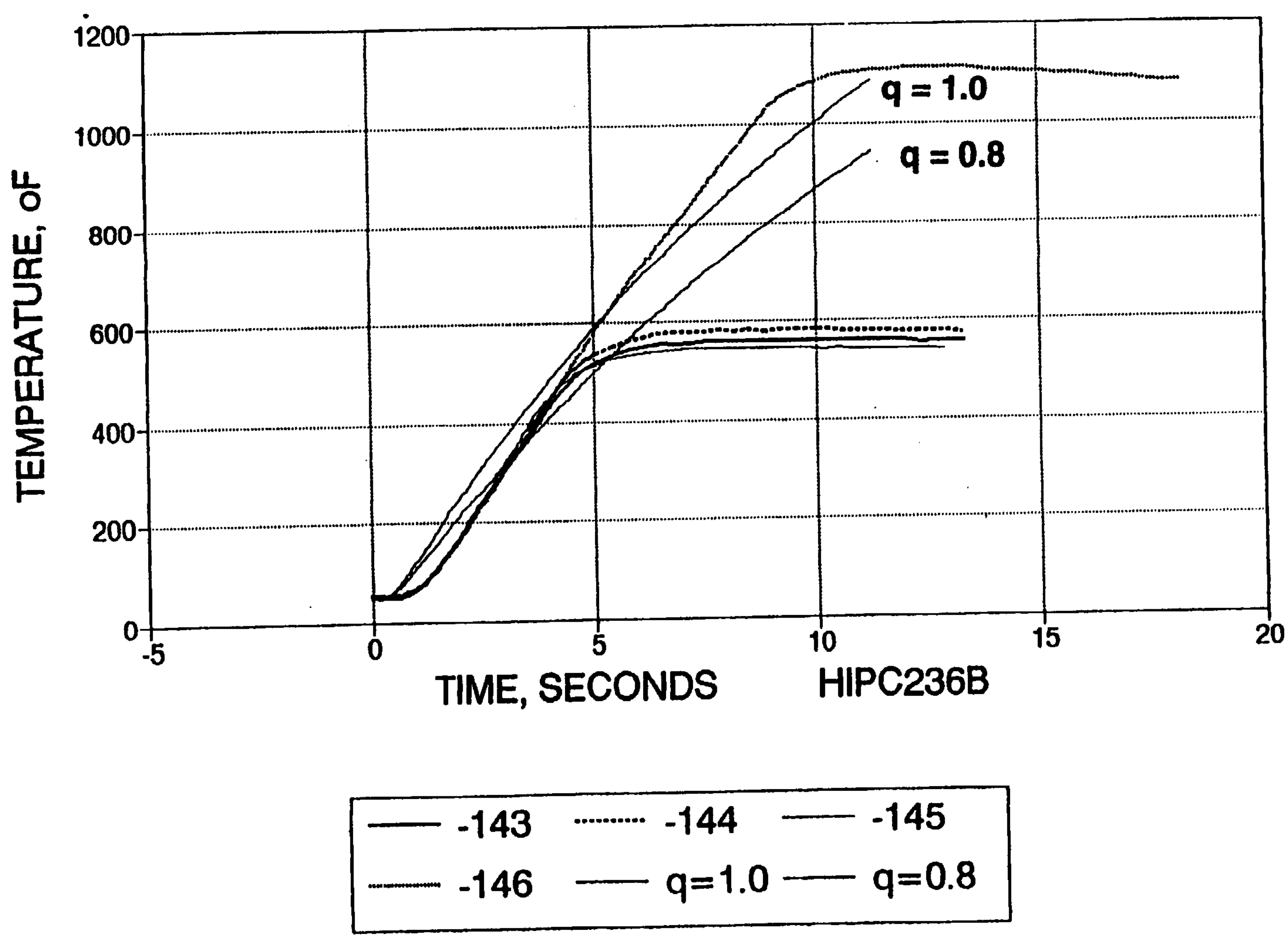
At Pc = 246 psia Cold Wall Heat Flux to the Chamber Is About 1.0 BTU (in^2 - sec)



7.81.E8.1

High Pressure Earth Storable Rocket Technology Program

Hot Wall Response
TC-3



High Pressure Earth Storable
Rocket Technology Program

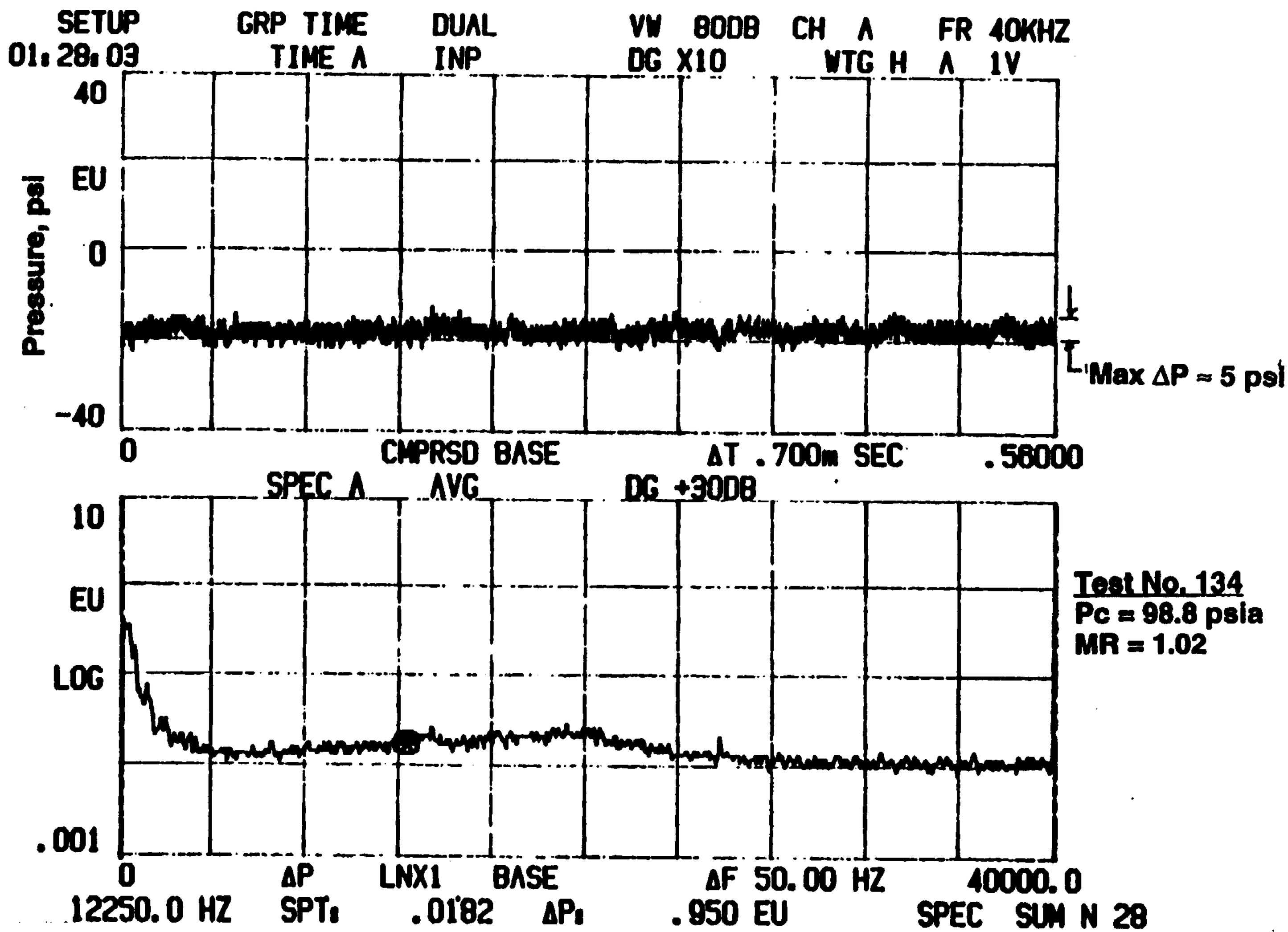
Chamber Thermal Management Conclusions

o Chamber wall heat flux is relatively low (ca. 1 btu/sec-in ^ 2)

o Chamber temperatures will be below operating limit, even at 500 psia

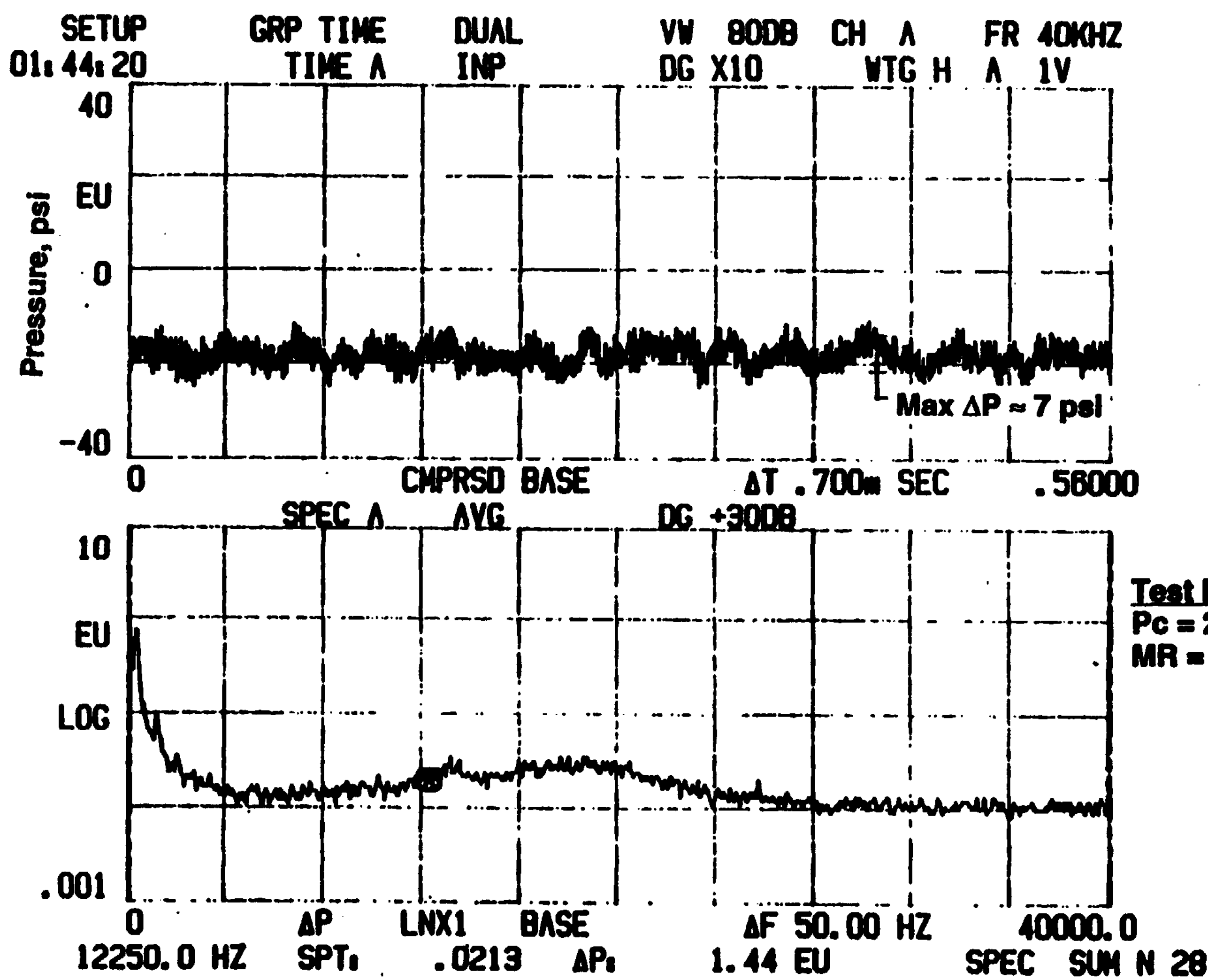
High Pressure Earth Storable Rocket Technology Program

FFC Testbed Chamber High Frequency Test -134



High Pressure Earth Storable Rocket Technology Program

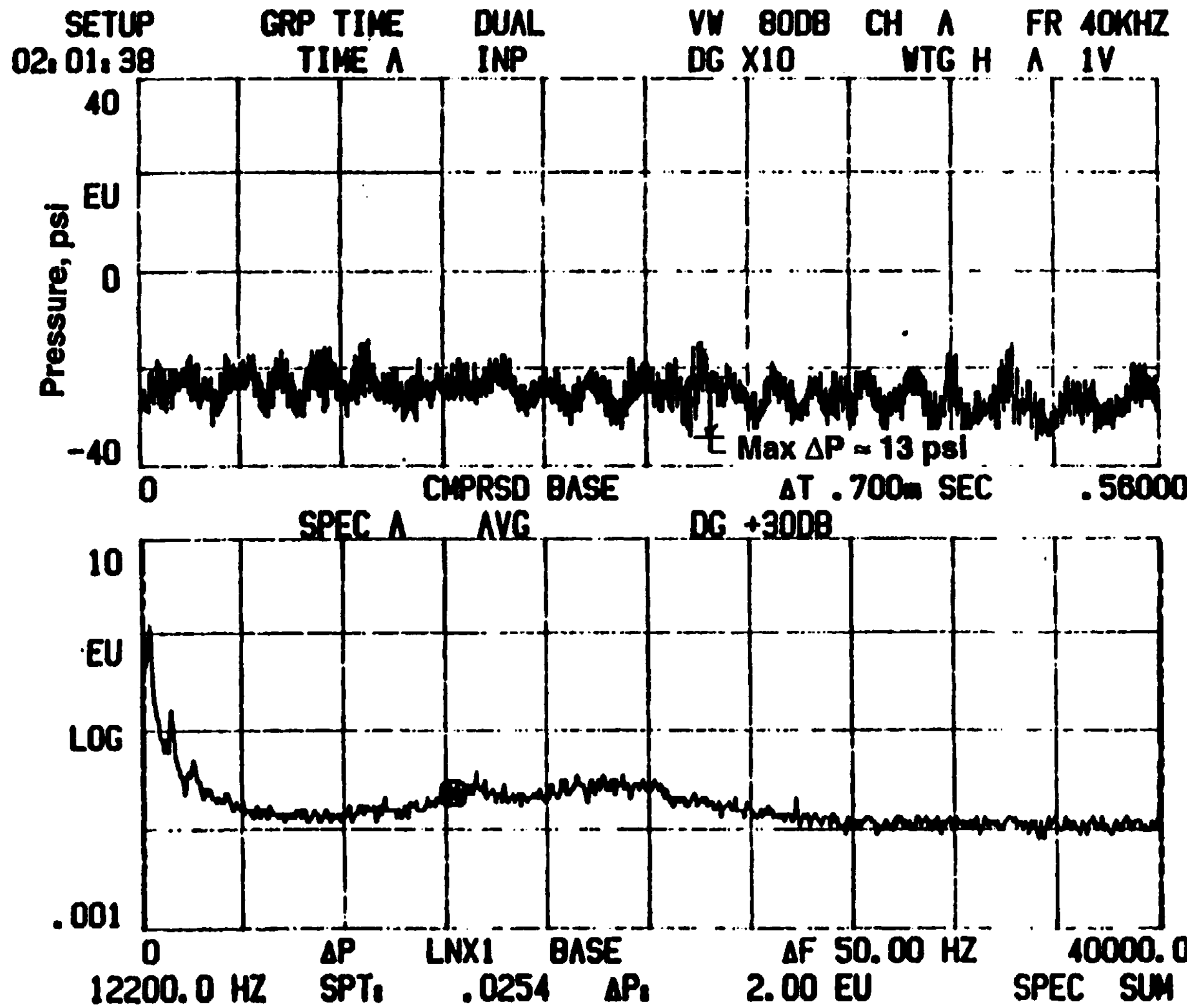
FFC Testbed Chamber High Frequency Test -138



Test No. 138
Pc = 247 psia
MR = 1.08

High Pressure Earth Storable
Rocket Technology Program

FFC Testbed Chamber High Frequency
Test -141

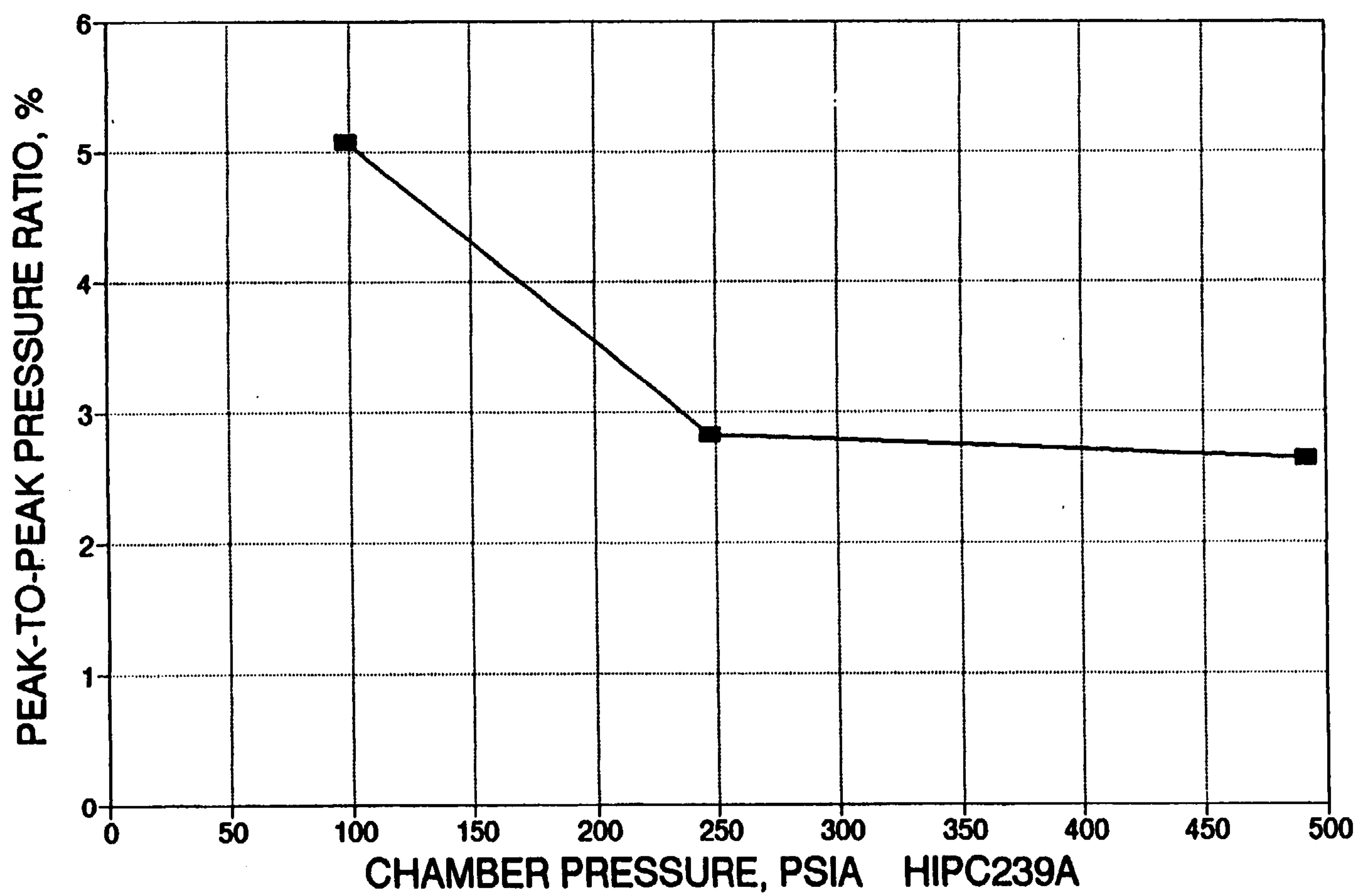


Test No. 141
Pc = 492 psia
MR = 1.05

Task 2 tests showed no instabilities in either chug or acoustic modes. Peak-to-peak chamber pressure fluctuations were 5% or less (typical production engine specifications are 3.5% for the LEROS 1 and 12% for the R4-D). Note that the relative magnitude of the chamber pressure fluctuations decreased as Pc increased. The low frequency, not quite organized, signal noted at 500 Pc had a frequency of roughly 40 Hz. This is too low to be chug; calculations show it could be an interaction between the PDFM piston stiction and the chamber.

High Pressure Earth Storable Rocket Technology Program

FFC Testbed Pc Variations Chamber High Freq



High Pressure Earth Storable
Rocket Technology Program

TASK 4 TESTBED--DESIGN

High Pressure Earth Storable Rocket Technology Program

Task 4 Testbed Design

- Design Philosophy
  - Tailor Design for N_2O_4/N_2H_4, e.g., Injector ΔP
  - Utilize 100-lbf Designs/Hardware to Take Advantage of the Previous Related Technology Contracts
- Provide Test Hardware to Conduct Tests Over a Range of Pc and MR Parameters
- Provide Flexibility in the Hardware Designs to Allow Different Assembly Configurations to be Tested Which Will Establish Critical design Criteria and Optimization of Parameters Relative to Performance, Thermal Characteristics, Stability, and Durability

17.130-4/20

High Pressure Earth Storable Rocket Technology Program

Design, Fabrication and Testing for Task 4

- Focus on Film Cooled With High Trip
- Platinum Trip and Rhenium Chamber to Withstand Thermal Environment
- Remain Flexible to Achieve Maximum Results
  - Unique Design Allows for Easy Change of Trip to Test Various heights
  - Chambers With Different Throat Sizes Utilized for Different Pressures
  - Task 2 Copper Throats Also Available and Usable if Necessary

17.130-4b/rt/11

- **Testbed Configuration – Description**

The injector pattern will be similar to the S/N 06 injector that was successfully tested for a cumulative duration of 6.2 hours. The only design change will be to adjust the metering orifices to obtain the target ΔP of 60 psia at nominal flowrate. The manifold will be the same and will include the 16 film coolant ports from which the film coolant can be adjusted with fittings with different orifice sizes.

The chamber will be an all rhenium design which will permit test durations much greater than the 10 seconds maximum of Task 2 to be conducted. This configuration will provide more realistic thermal characteristics which will be necessary input for the Option 1 hardware design. As a minimum, chambers will be procured for testing at Pc's of 100 and 250 psia.

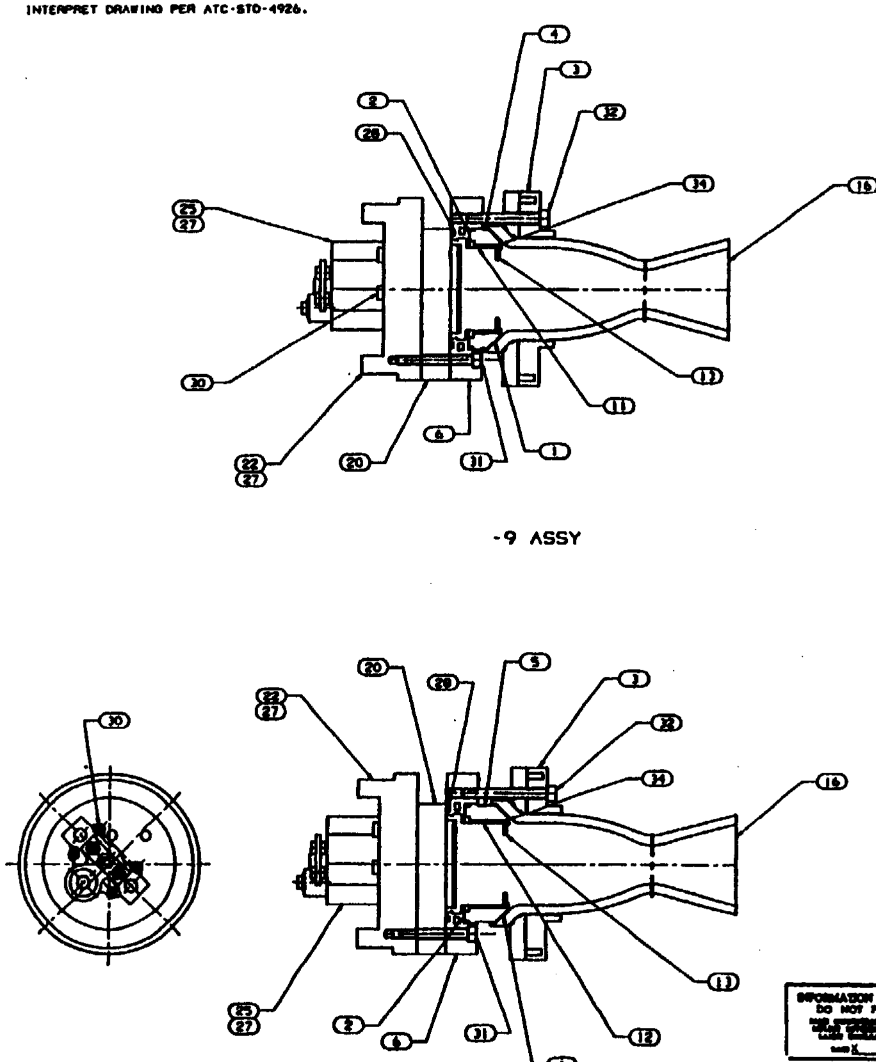
The trip will be fabricated from platinum to withstand the high temperatures and oxidation environment over long durations. Trip heights will be .05 in. and .25 in. respectively for test Pc's of 100 and 250 psia. The trip housing will control the trip distance from the injector face; 0.55 and 0.75 in. lengths will be available for testing.

High Pressure Earth Storable Rocket Technology Program

HIPC Engine

NOTES:

1 INTERPRET DRAWING PER ATC-STD-4926.



-19	-9						
3	3		28700-90F -1	SCREW			34
8			20097-3F -28	SCREW			33
	8		20097-3F -24	SCREW			32
7	7		20097-9BC -22	SCREW			31
4	4		20097-94C -6	SCREW			30
							29
1	1		2-140	SEAL	PARKER		28
4	4		AR10304 -012A0	SEAL			27
							26
1	1			VALVE	MOOG		25
							24
							23
1	1		A-5364-030	ADAPTER, VALVE			22
							21
1	1		1208178-9	INJECTOR			20
							19
			1208177-3	CHAMBER			18
1	1		1208177-2	CHAMBER			17
			1208177-1	CHAMBER			16
			1208176-3	TRIP			15
1	1		1208176-2	TRIP			14
			1208176-1	TRIP			13
1			1208174-3	HOUSING, TRIP			12
	1		1208174-2	HOUSING, TRIP			11
			1208174-1	HOUSING, TRIP			10
							9
							8
							7
1	1		1207296-9	ADAPTER			6
1			1207294-2	RING			5
	1		1207294-1	RING			4
1	1		1199065-9	CLAMP			3
1	1		-2	SEAL	GRAFOIL		2
2	2		-1	SEAL	GRAFOIL		1

DEVELOPMENT HARDWARE
FOR TEST OR EXPERIMENTATION

KNIGHT 94-03-2?

DO NOT SCALE DRAWING

ENGINE, HIPC

E 0582- 1208175

1/1

High Pressure Earth Storable Rocket Technology Program

Testbed Configuration

- Description

Component	P/N	Description
• Valve	Model 53X186	Gas Actuated, Solenoid Pilot Valve Controlled, Bipropellant Valve
• Injector	1209740	Will Be Similar to S/N 06 But With Fuel Film Cooling
• Chamber	1208177	Rhenium Chambers (No Iridium) With Different Throat Diameters
• Trip	1208176	Platinum Trip With Different Height Configuration
• Trip Housing	1208174	Provides Installation and Support for the Trip and Will Provide Choice of Trip Length From Injector Face
• Ring	1207294	Provides Installation and Support to the Trip/Housing Assembly
• Adapter	1207296	Forms the Resonator Cavity and Interface for the Injector and the Ring

17.130-4b/rt/4

High Pressure Earth Storable Rocket Technology Program

Injector Design

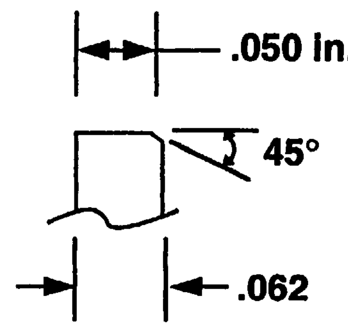
- Platelet Stack (P/N 1209740-9)
  - Pattern Will Be Similar to S/N 06 Injector
  - No. of Elements = 92 Ox-On-Fuel Doublets
  - Type Elements: Oxid and Fuel Are Splashplate Design
  - Target ΔP = 60 psi
  - Target % Fuel Film Cooling = 35%
  - Material: 347 CRES
- Manifold (P/N 1206357-9)
  - Same as S/N 05 and S/N 06-2
  - Will Have 16 each Ports for Fuel Film Cooling
  - Material 304L CRES
- Assembly (P/N 1208178)
  - Platelet Stack and Manifold Bonded Together

17.130-4b/rt/5

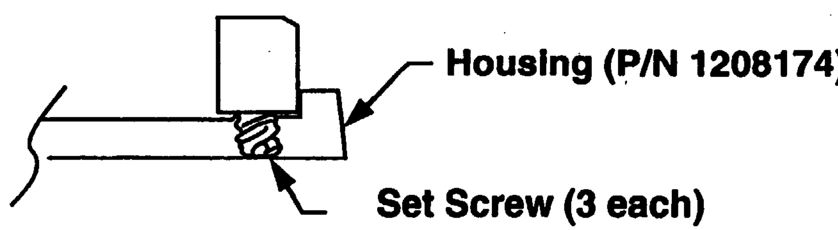
High Pressure Earth Storable Rocket Technology Program

Trip Design

- Configuration



- P/N 1208176
- Height: .05 in. for Tests at Pc = 100 psia
  .25 in. for Tests at Pc = 250 psia
  .40 in. for Tests at Pc = 500 psia
- Material: 90% Platinum + 10% Rhodium
- Installation:

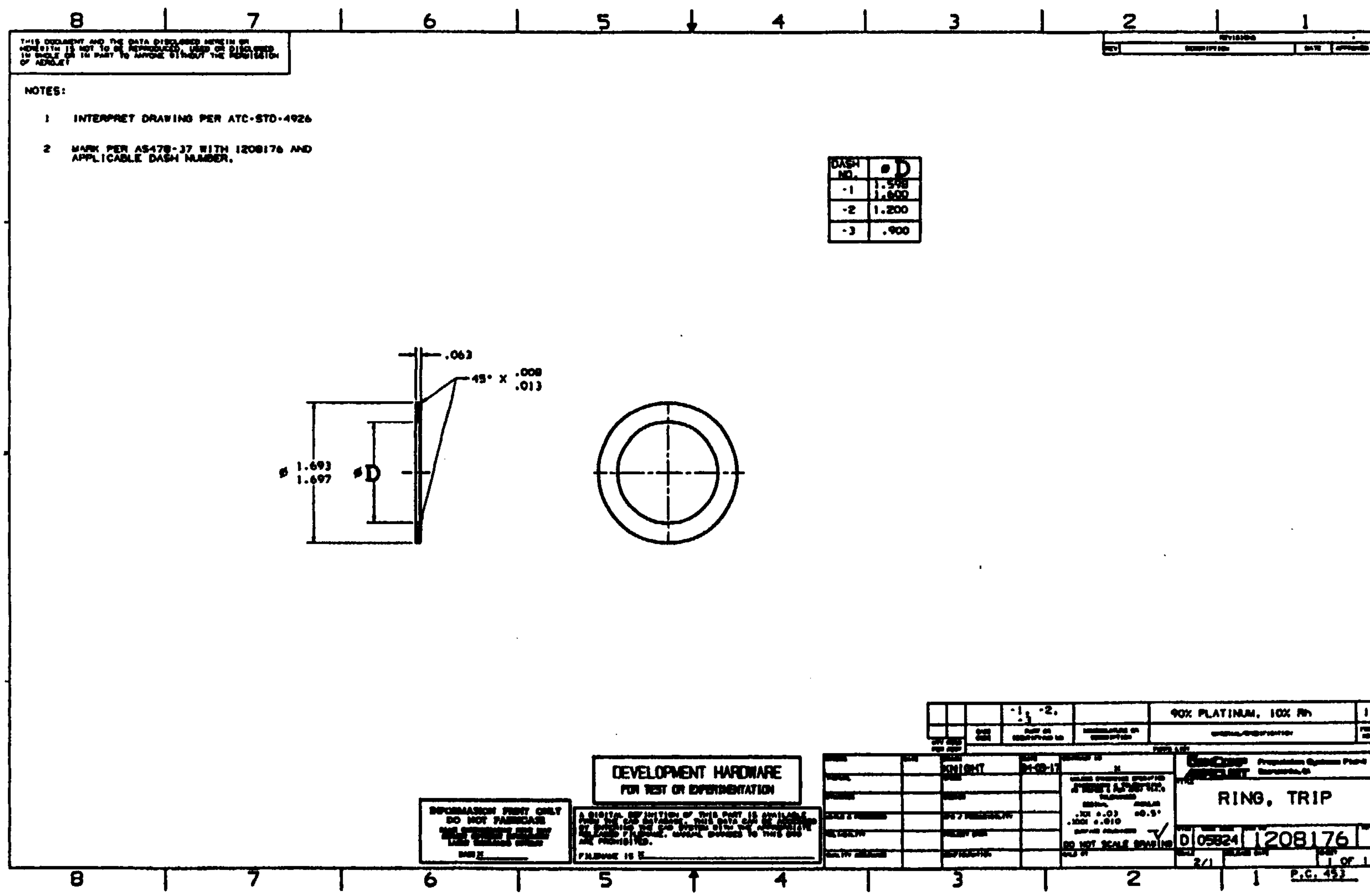


- Length to Injector Face:
  0.55 in. and 0.75 in.

7.81.ES.2

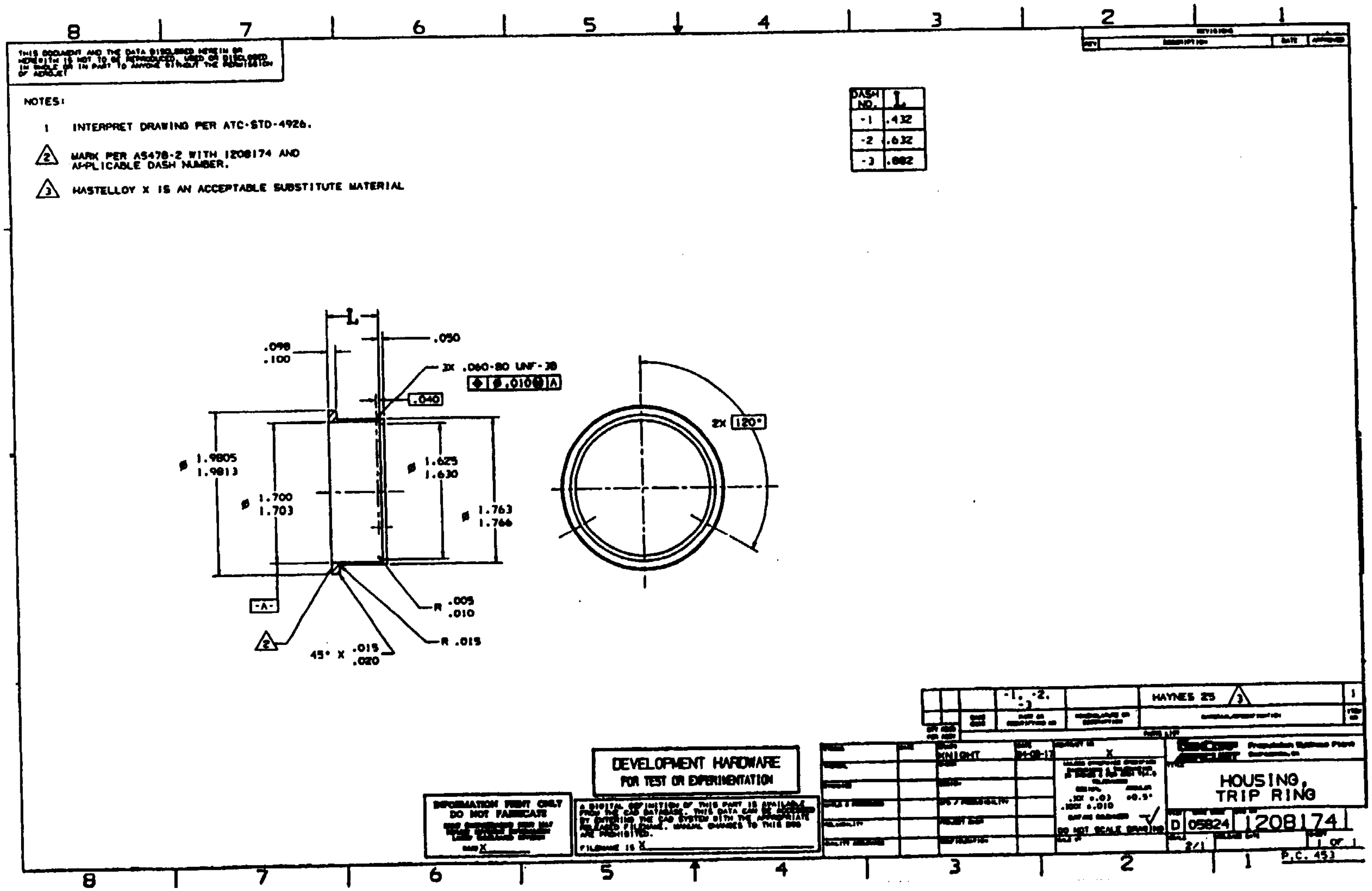
High Pressure Earth Storable Rocket Technology Program

Trip Ring



High Pressure Earth Storable Rocket Technology Program

Trip Ring Housing



High Pressure Earth Storable Rocket Technology Program

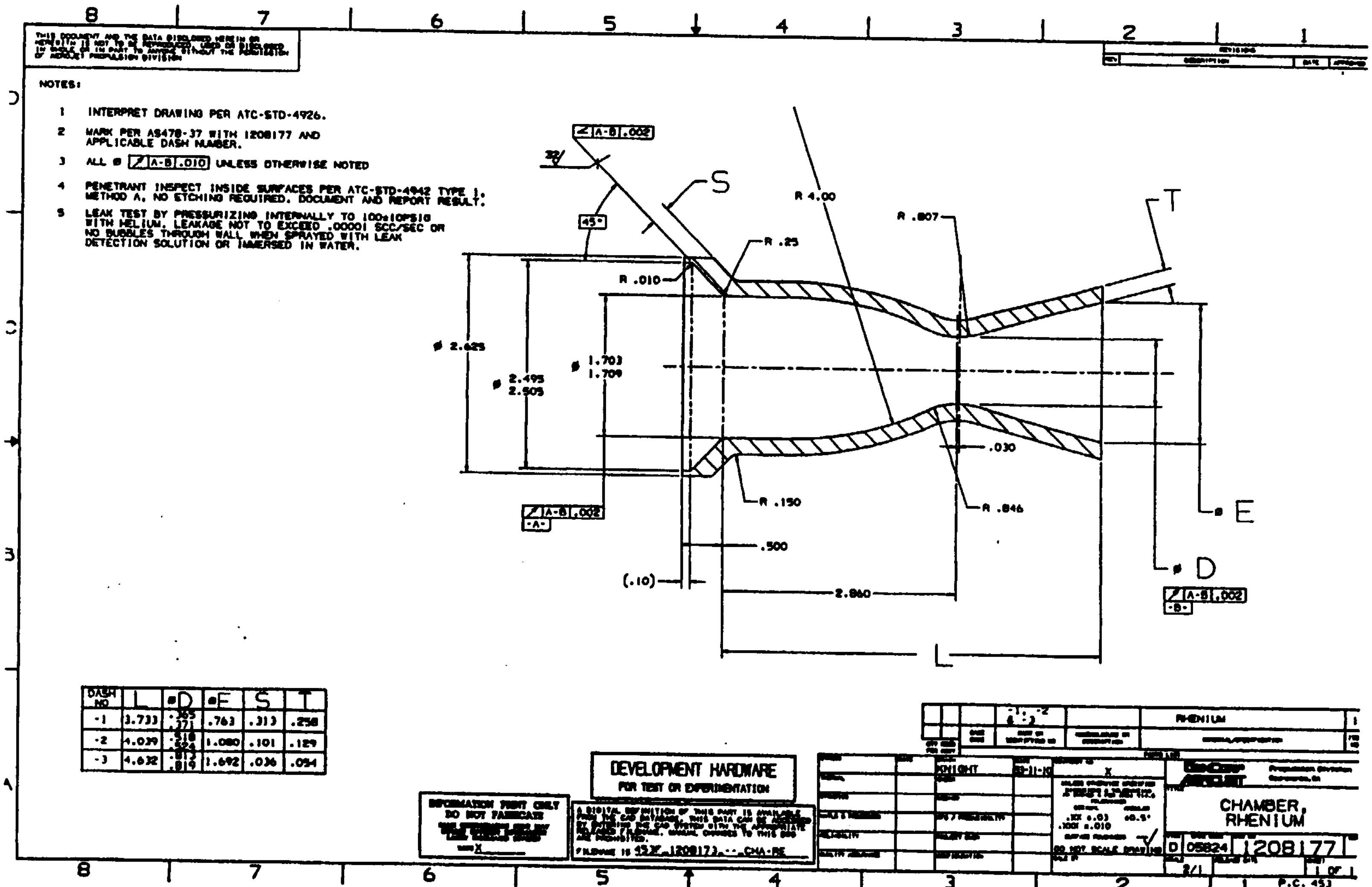
Task 4 Chamber: Design Requirements

- Have Adequate Structural Safety Factor: (Thick)
- Minimize Thermal Load to Front End (Thinner at Front, Thicker at Throat)
- Reduce Fabrication Cost (Thinner; Minimize Machining)
- Provide Adequate Test Life (Thicker)

17.130-4/22

High Pressure Earth Storable Rocket Technology Program

Rhenium Chamber



DASH NO	L	⌀D	⌀E	S	T
-1	3.733	.365 .371	.763	.313	.258
-2	4.039	.518 .524	1.080	.101	.129
-3	4.632	.813 .819	1.692	.036	.054

High Pressure Earth Storable Rocket Technology Program

Summary of Logic for Choice of Chamber Wall Thickness

Requirement	Location	Design Value	Required Thickness, in.		
			at Pc = 100	at Pc = 250	at Pc = 500
1. Structural Safety Factor	Chamber Cylinder	2.0	0.035	0.090	0.190
	Chamber Flange	2.0	0.017	0.062	0.246
2. Thermal Margin	Flare	Meet Margin	0.017	0.062	0.246
	Nozzle	Maintain Thickness at Chamber Value	0.035	0.035	0.190
3. Test Life – Erosion Hardware	Chamber	Based on 14 lb Data	0.019	0.039	0.067
4. Ease of Fabrication	Throughout	Minimize Machining			
Resulting Dimension:	Flare		0.036	0.101	0.313
	Chamber		0.054	0.129	0.258
	Throat		0.054	0.129	0.258
	Nozzle		0.054	0.129	0.258

17.130-4/23

High Pressure Earth Storable Rocket Technology Program

- <u>Mechanical</u>
  - <u>Interfaces</u>
    - Seals

      Grafoil Material Is Used for Hot Gas Seals at 2 Joints: (1) Adapter-to-Ring and (2) Ring-to-Chamber

    - Assembly

      The Chamber, Ring, and Trip Can Be Removed Without Removing the Valve/Injector From the Test Stand. The Chamber Assembly Bracket Also Forms the Manifold for the Hydrogen Blanket That Is Necessary to Preclude Oxidation of the Rhenium Chamber During Firing

17.130-4b/rt/6

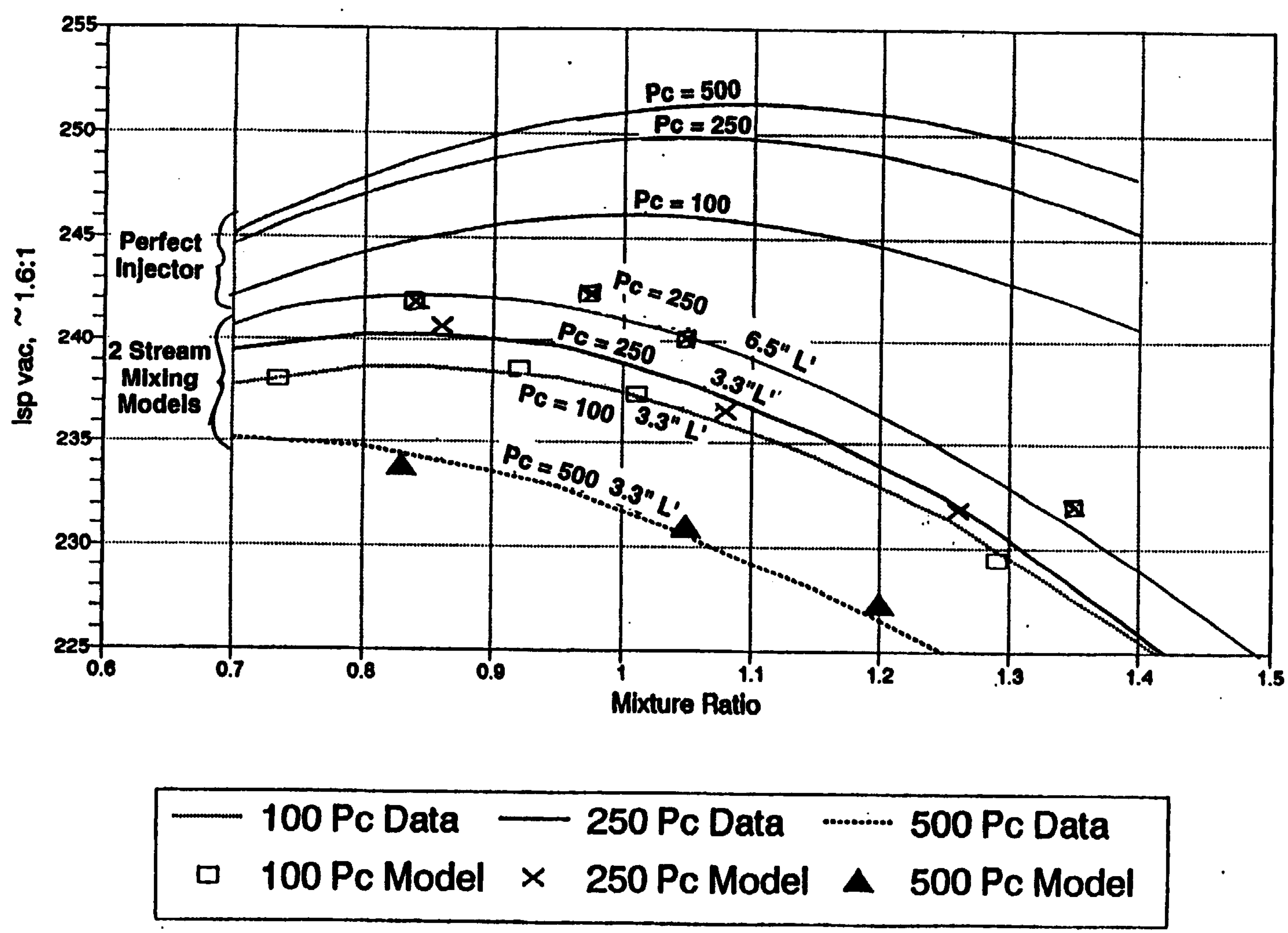
ANALYSES SUPPORTING DESIGN

- **Calibration of 2 Stream Model to the 1.6:1 Data**

A simple 2 stream mixing model was calibrated to the fuel film cooled low area ratio data in order to extrapolate the performance to the 300:1 area ratio nozzle. This extrapolation method was chosen over using a constant injector ERE for the two area ratios because the shape of the ERE curve with mixture ratio indicated a significant mixture ratio maldistribution loss. Since the variation of Isp with mixture ratio changes significantly from 1.6:1 area ratio to 300:1, e.g., the peak of the Isp curve shifts to higher mixtures with increased area ratio, it was determined that a 2 stream tube Em model would be a more accurate method for the extrapolation. A variable Em profile with mixture ratio was generated to match the 100 psia Pc data. This profile was then used for all the other data with a constant factor applied to either raise or lower the entire curve to match the level of the measured performance. This Em profile matched all of the mixture ratio trends of the different chamber pressures and chamber lengths tested, as shown in the figure. These Em profiles were then used to predict the performance at the 300:1 area ratio, as shown in the following chart.

High Pressure Earth Storable Rocket Technology Program

100 Lbf Fuel Film Cooled Data
Calibration of a 2-Stream Model

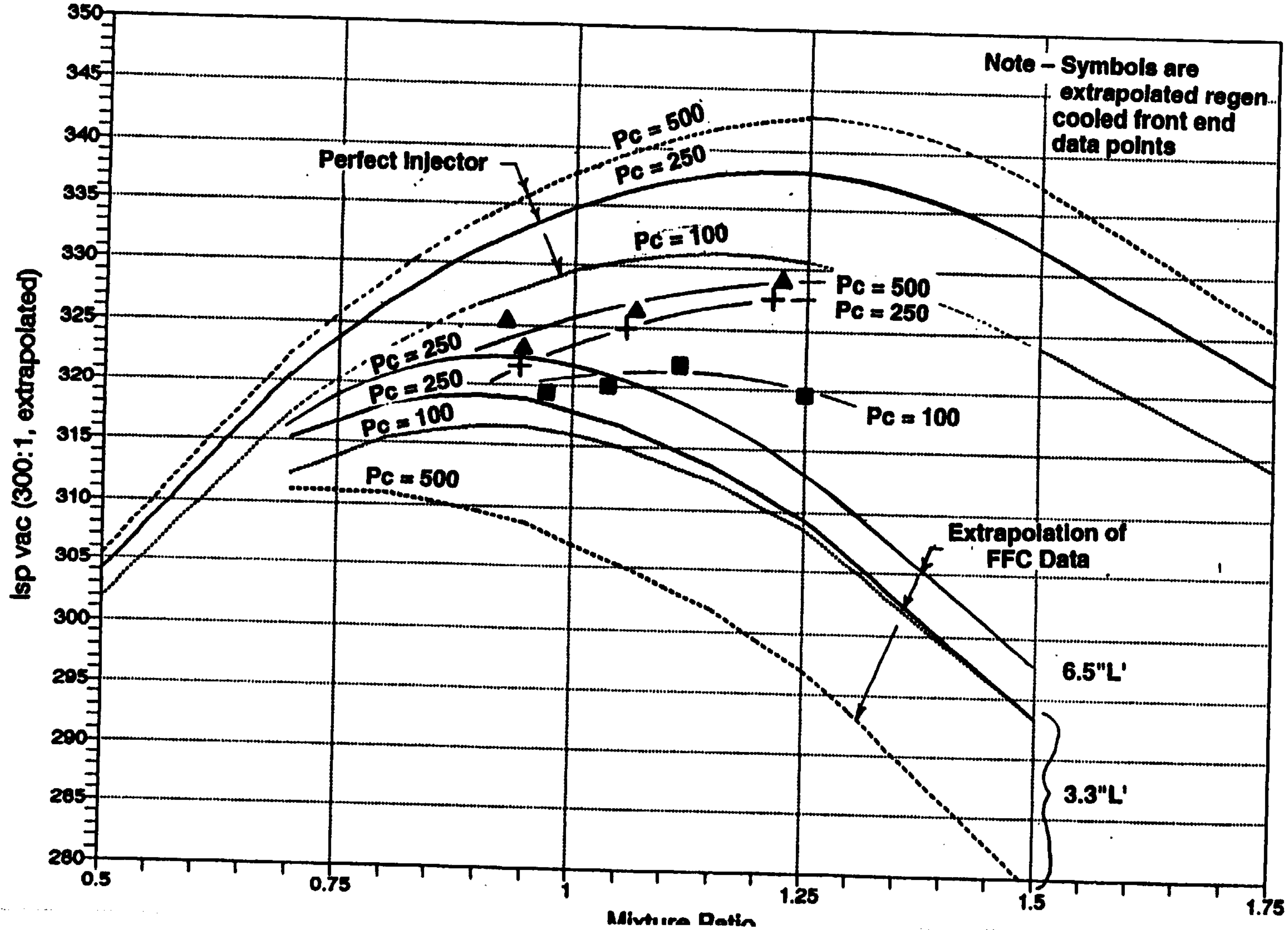


- **Extrapolation of Low Area Ratio Performance to 300:1**

The calibrated Em model for the fuel film cooled low area ratio performance was used to predict the 300:1 performance for that injector in this chart. The extrapolation for the three different chamber pressures are shown as well as the data at the 250 psia Pc with the added L' section. The data from the regeneratively cooled front end is also shown as data points on this chart. The individual data points for this data set were extrapolated to the 300:1 area based on the calculated injector efficiency (ERE) from the individual low area ratio tests. The predicted 300:1 performance for the regen cooled front end data is higher than that for the fuel film cooled injector. The higher mixture ratio maldistribution loss for the fuel film cooled injector causes its injector efficiency to drop as the mixture ratio approaches the peak in the theoretical Isp curve, shown in the figure. The regen front end engine has an injector efficiency which is nearly constant over the tested mixture ratio range, evidence of a lower mixture ratio maldistribution loss. Consequently this injector is able to maximize Isp consistent with the theoretical curve.

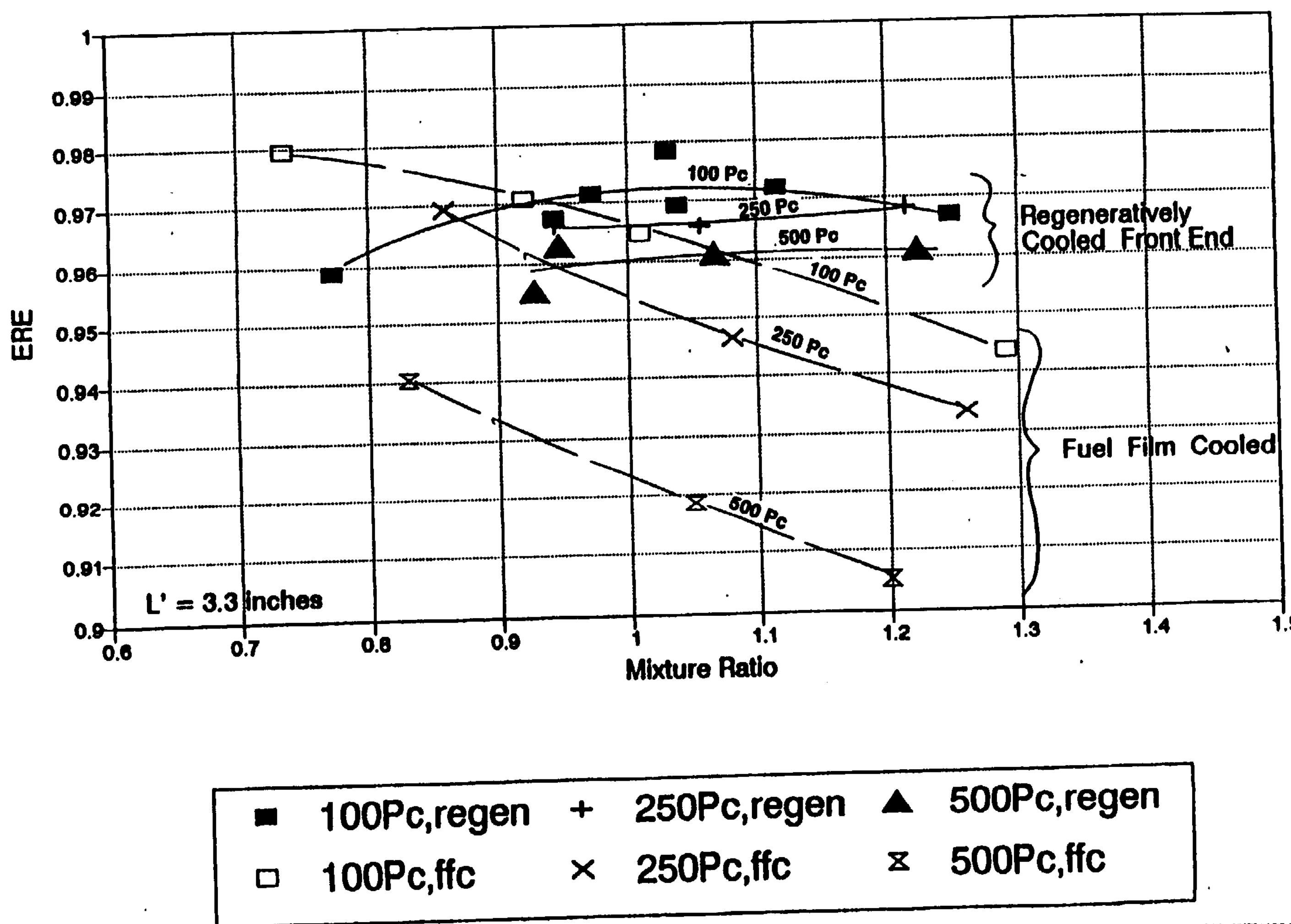
High Pressure Earth Storable Rocket Technology Program

300:1 Extrapolation of Regen Cooled Front End and FFC Data



High Pressure Earth Storable Rocket Technology Program

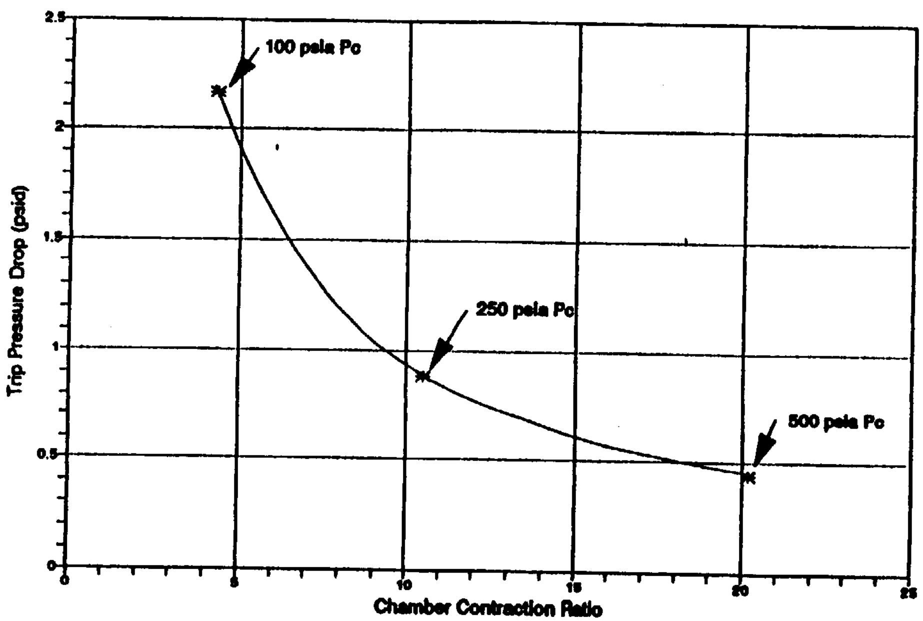
100 Lbf Data, Pc from 100 psia to 500 psia Regen and 42% FFC Data



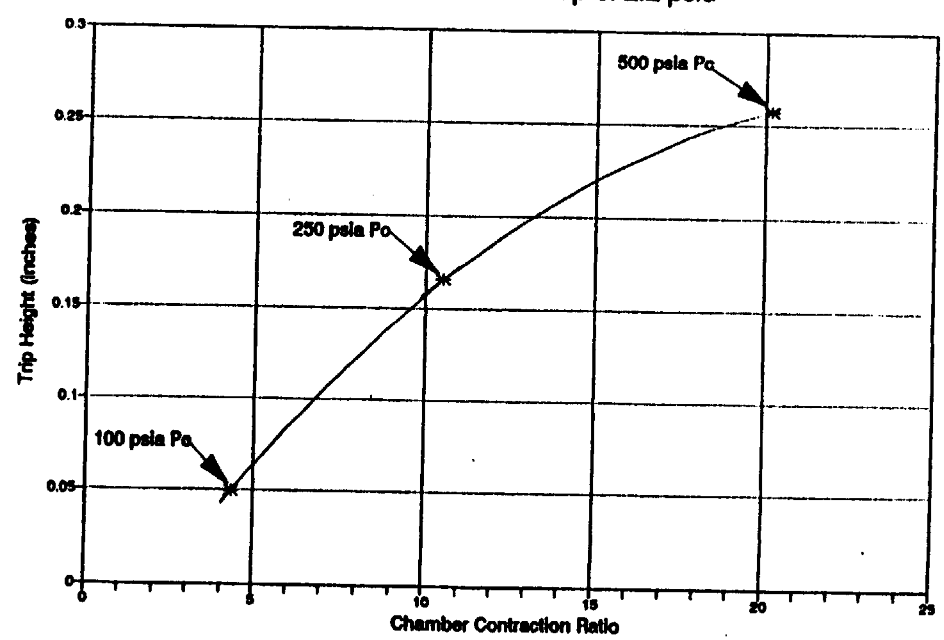
High Pressure Earth Storable Rocket Technology Program

Task 4 Trip Is Designed to Provide Good Mixing of FFC With Core Flow

Chamber Trip Pressure Drop vs CR
.05" trip



Required Chamber Trip Height
for Constant Pressure Drop of 2.2 psid

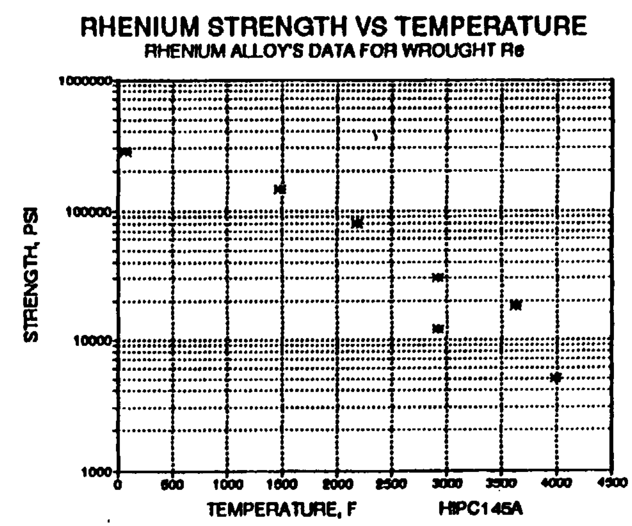


HIPC145

11-3-93

RHENIUM STRENGTH DATA VS TEMPERATURE

MATERIAL	TEMP F	Sult PSI	CONDITION
WROUGHT R	68	280,000	
	1472	145,000	
	2192	80,000	
	2912	30,000	
	3632	18,000	
	2912	12,000	2.24 HR TO RUPTURE/6%
	3992	5,000	0.91 HR TO RUPTURE/8%



HIPC144

TASK 4.0 CHAMBER STRESS ANALYSIS

CHAMBER WALL--RHENIUM

S=PD/2t

P PSIA	WORST CASE D1 IN	D2 IN	Davg IN	t wall IN	S PSI	MAX. TEMP oF	0.91 HR RUPTURE Sr, PSI	SAFETY FACTOR ON Sr	CORRO-SION ALLOW., IN	DESIGN THICK. IN
100	1.709	1.779	1.744	0.035	2491	4000	5000	2.01	0.019	0.054
150	1.709	1.815	1.762	0.053	2493	4000	5000	2.01	0.026	0.079
200	1.709	1.853	1.781	0.072	2474	4000	5000	2.02	0.032	0.104
250	1.709	1.890	1.800	0.090	2485	4000	5000	2.01	0.039	0.129
400	1.709	2.010	1.860	0.150	2471	4000	5000	2.02	0.056	0.207
500	1.709	2.090	1.900	0.190	2493	4000	5000	2.01	0.067	0.258
600	1.709	2.180	1.945	0.236	2477	4000	5000	2.02	0.078	0.313
750	1.709	2.320	2.015	0.305	2473	4000	5000	2.02	0.093	0.398
1000	1.709	2.580	2.145	0.436	2462	4000	5000	2.03	0.117	0.552

CHAMBER FLARE

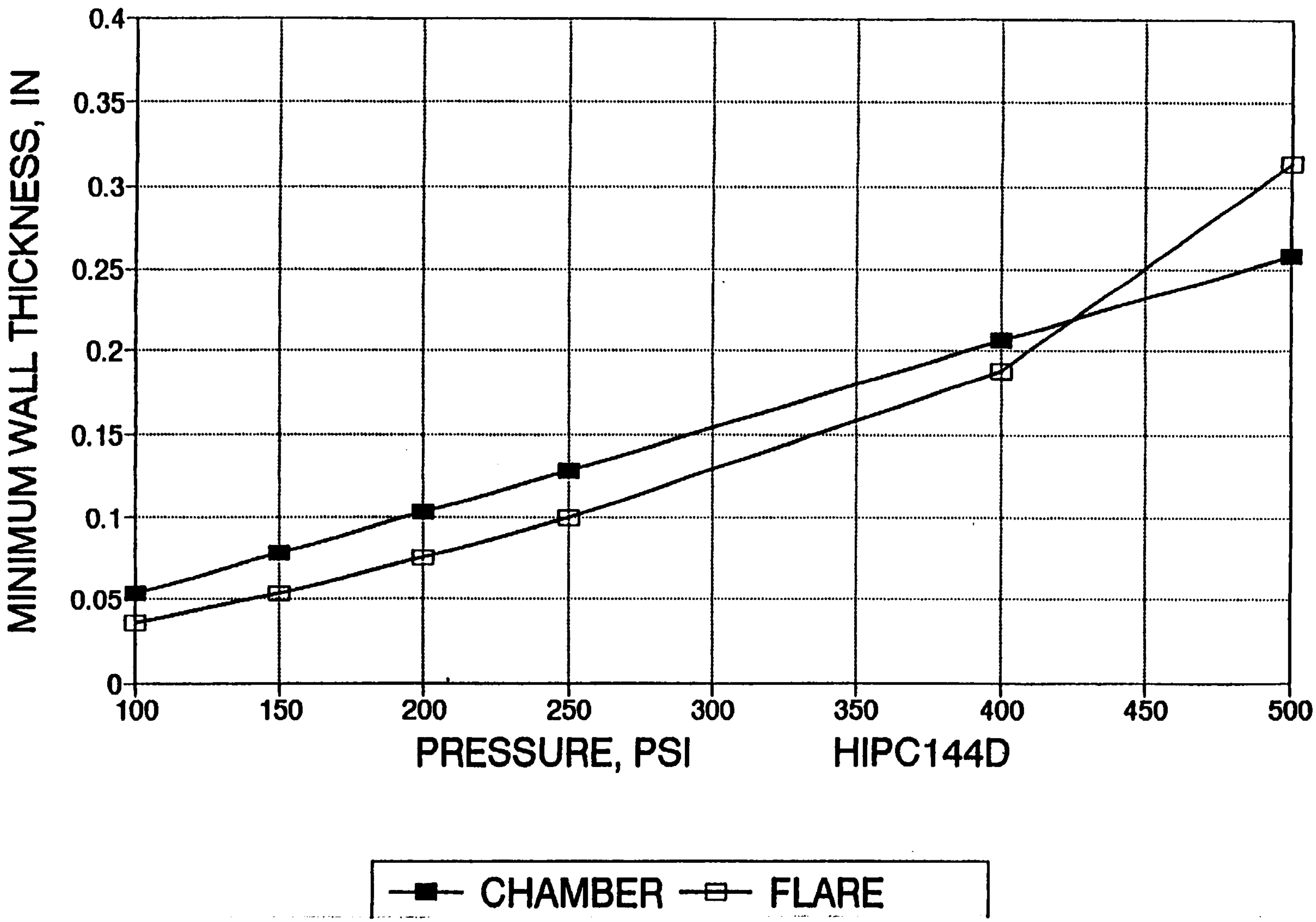
RHENIUM

=D*Pc*k/2t

CONICAL SEAL AVG. DIA.	Pc PSIA	L IN	t wall IN	r IN	L/t	r/t	APPROX. STRESS CONC., k	T oF	RUPTURE STRENGTH 2.2 HR psi	APPROX STRESS psi	SAFETY FACTOR	CORRO-SION ALLOWANCE IN	DESIGN THICK. IN
2.167	100	0.35	0.0173	0.015	20.231	0.867	1.25	3000	16000	7829	2.04	0.019	0.036
2.167	150	0.35	0.028	0.015	12.500	0.536	1.35	3000	16000	7836	2.04	0.026	0.054
2.167	200	0.35	0.044	0.015	7.955	0.341	1.6	3000	16000	7880	2.03	0.032	0.076
2.167	250	0.35	0.062	0.015	5.645	0.242	1.8	3000	16000	7864	2.03	0.039	0.101
2.167	400	0.35	0.132	0.015	2.652	0.114	2.4	3000	16000	7880	2.03	0.056	0.188
2.167	500	0.35	0.246	0.015	1.423	0.061	3.6	3000	16000	7928	2.02	0.067	0.313
2.167	600	0.35	0.312	0.015	1.122	0.048	3.8	3000	16000	7918	2.02	0.078	0.390
2.167	750	0.35	0.43	0.015	0.814	0.035	4.2	3000	16000	7937	2.02	0.093	0.523
2.167	1000	0.35	0.615	0.015	0.569	0.024	4.5	3000	16000	7928	2.02	0.117	0.732

High Pressure Earth Storable Rocket Technology Program

Re Chamb. Wall, Inc. Corros. Allow.
T=4000F in Cyl, 3000F in Flare; SF=2.0

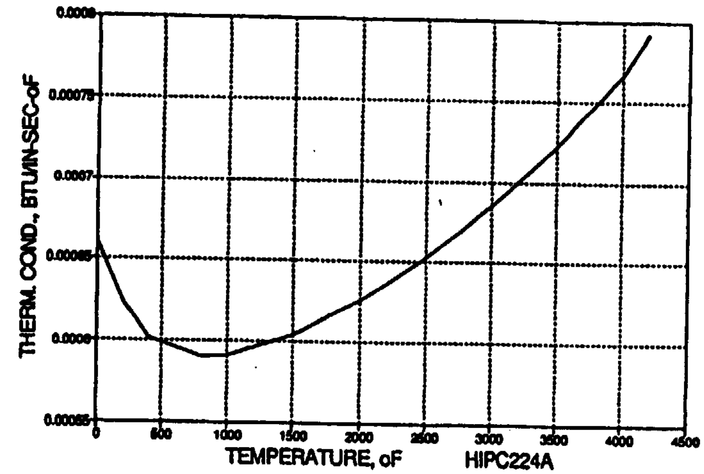


HIPC224 RHENIUM THERMAL PROPERTIES
3-20-94 VS TEMPERATURE

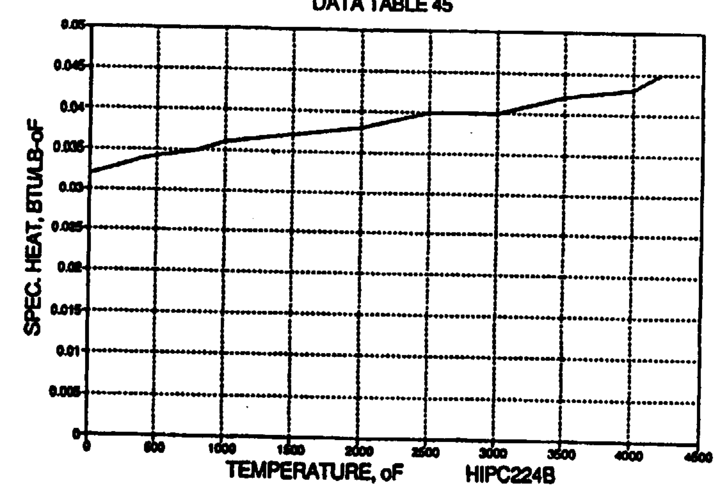
T, oF	T, oK	k, BTU/ SEC-IN-o [1]	Cp, BTU/ LB-oF [2]
0	255	0.00066	0.032
200	366	0.000623	0.033
400	477	0.000602	0.034
800	700	0.00059	0.035
1000	811	0.000591	0.036
1500	1089	0.000604	0.037
2000	1366	0.000626	0.038
2500	1644	0.000653	0.04
3000	1922	0.000685	0.04
3500	2200	0.000722	0.042
4000	2477	0.000766	0.043
4200	2589	0.000792	0.045

[1] TPRC DATA/WRT OCT 86
[2] DATA TABLE 45

RHENIUM–THERMAL CONDUCTIVITY
TPRC DATA



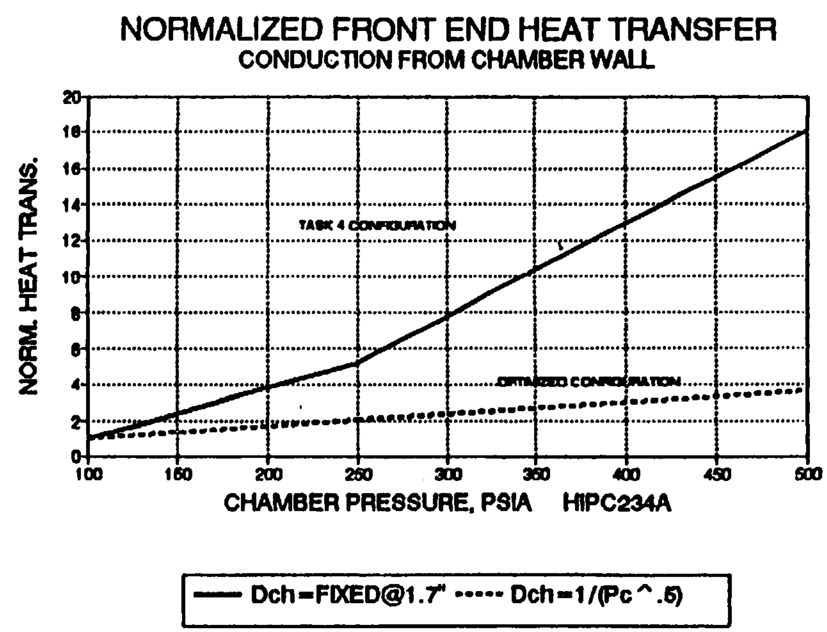
RHENIUM–SPECIFIC HEAT
DATA TABLE 45



High Pressure Earth Storable Rocket Technology Program

Heat Transfer Comparison

Pc	t	Dc	FRONT END Q' [RELATIVE]	FRONT END Q' NORMALIZED	COMMENT
100	0.032	1.700	0.054	1.0	FIXED Dc
250	0.080	1.700	0.283	5.2	"
500	0.160	1.700	0.986	18.1	"
100	0.032	1.700	0.054	1.0	Dc=1/(Pc)^0.5
250	0.051	1.075	0.113	2.1	"
500	0.072	0.760	0.197	3.6	"



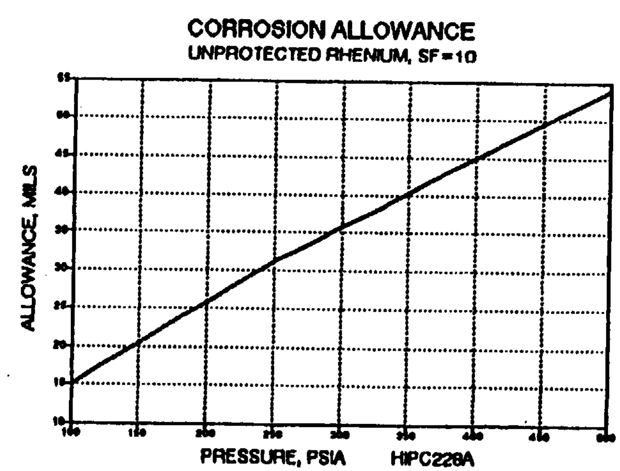
HIPC228
3-21-94

RHENIUM OXIDATION RATE

From 14# testbed testing, nominal erosion rate of unprotected rhenium is 10 mil/hour. Required test time of Task 4 Re chambers is about 10*60=600 sec or 0.167 hour, or 1.67 mil erosion. Providing a safety margin of 10 gives an erosion allowance of 16.7 mils (round up to 20 mils) to account for uncertainties in the data, local concentration, and to provide a safe factor on testbed life.

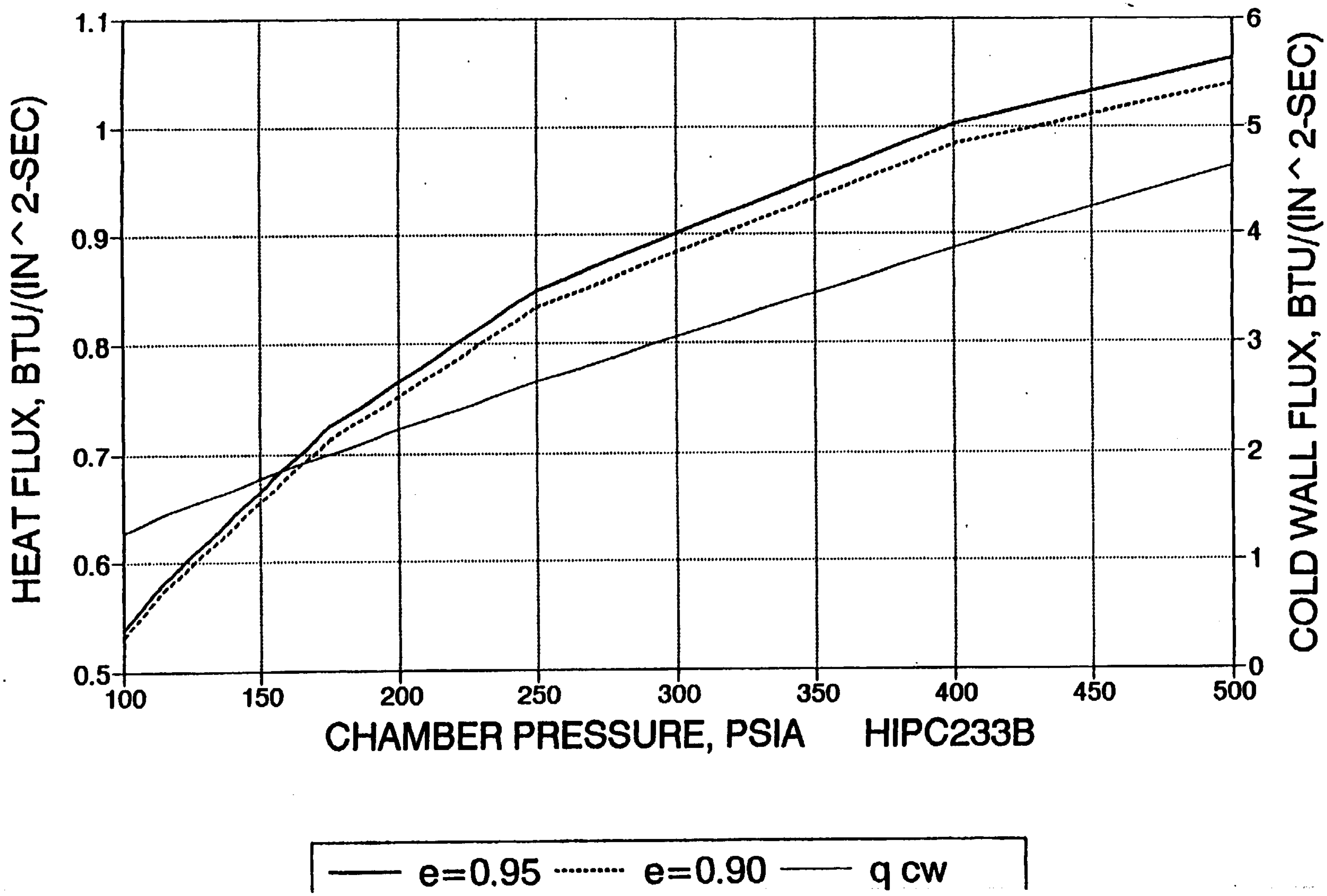
There are no test data presently available on the effect of pressure on rhenium mass loss rate. Assuming mass loss is analogous to heat transfer, the rhenium erosion rate should be proportional to P ^ 0.8. Therefore, the chamber thickness required to provide a safe operating margin for stress has been increased by a corrosion allowance.

PRESSURE, PSIA	SAFETY FACTOR	Re LOSS RATE, mil/hr	DESIGN LOSS RATE mil/hr	DESIGN LIFE HR	CORROSION ALLOW. mil
100	10	8.94	89.4	0.167	14.9
115	10	10.00	100.0	0.167	16.7
250	10	18.61	186.1	0.167	31.1
500	10	32.41	324.1	0.167	54.1



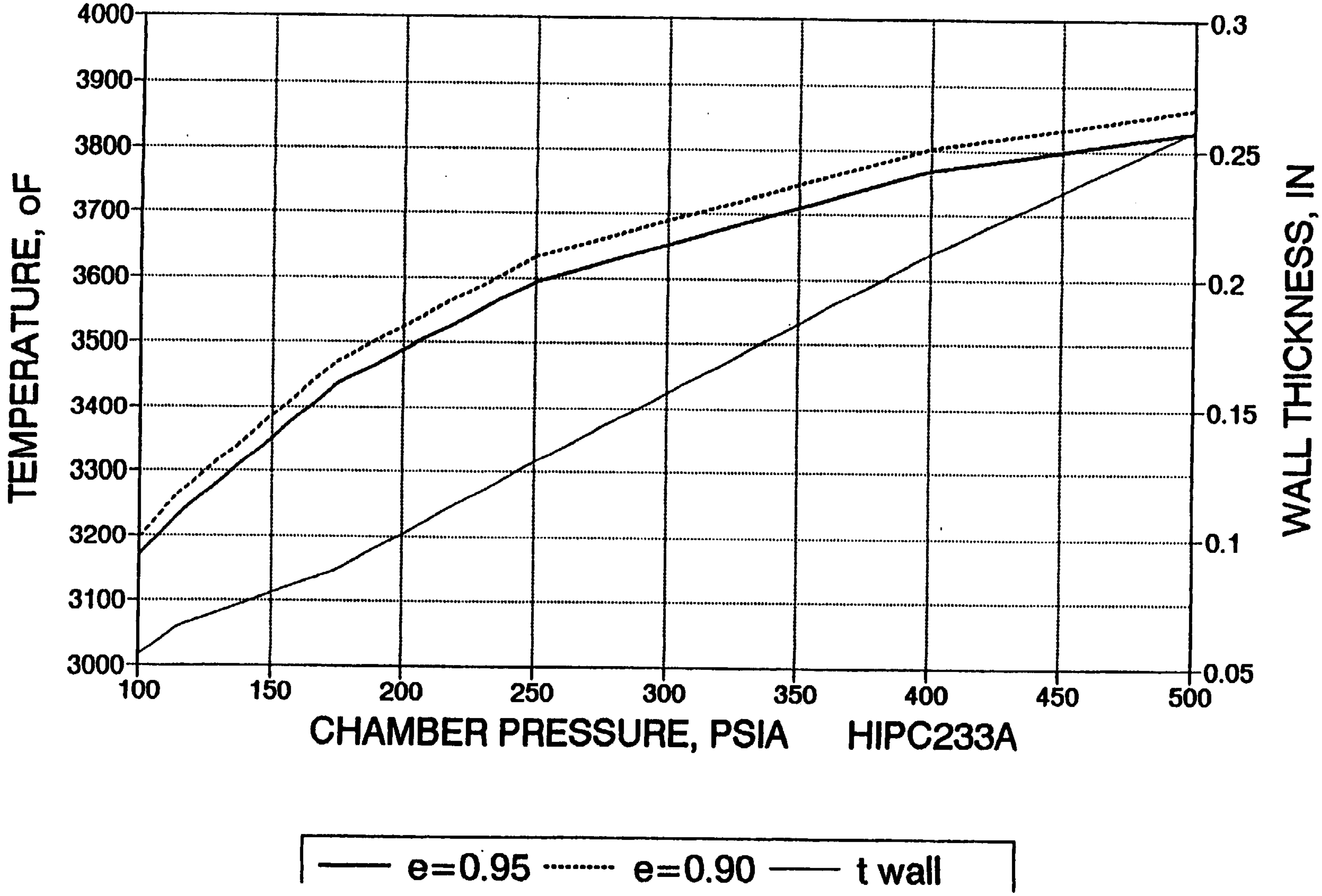
High Pressure Earth Storable Rocket Technology Program

Rhenium Chamber Heat Transfer
Calc. for hg =0.000263 at Pc = 115 psia



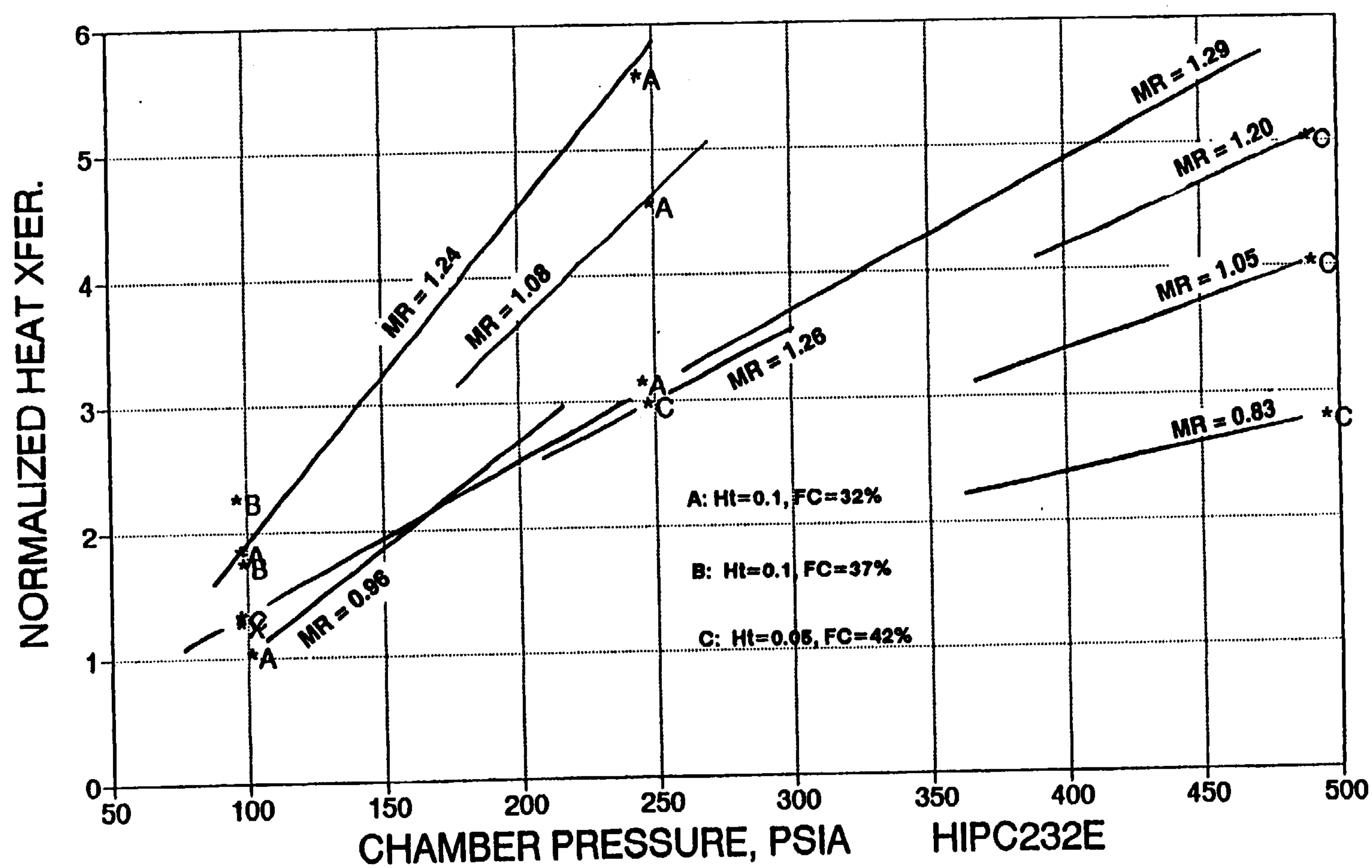
High Pressure Earth Storable Rocket Technology Program

Rhenium Chamber Temperature
Calc. for hg =0.000263 at Pc = 115 psia



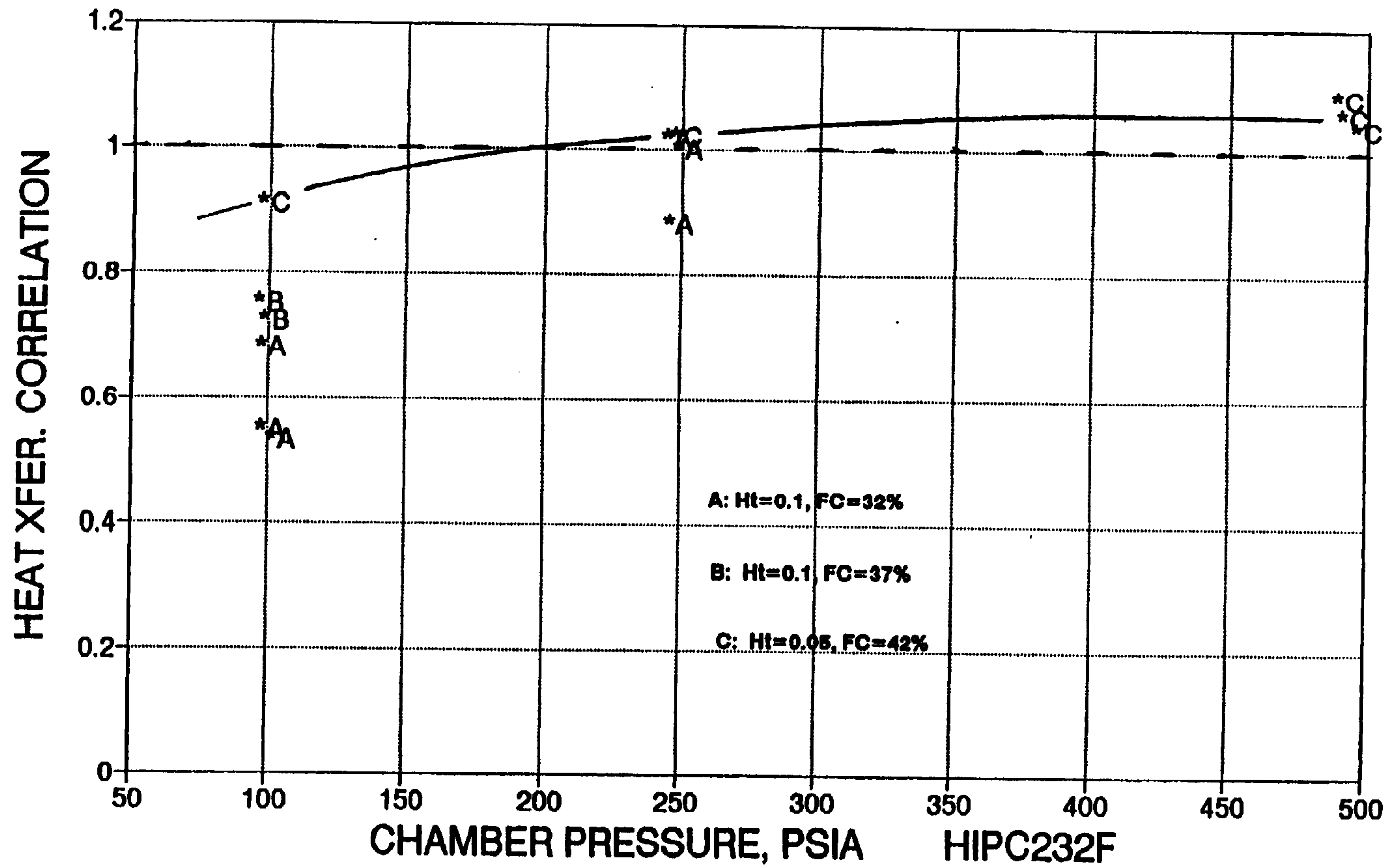
High Pressure Earth Storable Rocket Technology Program

Normalized Trip Heat Flux
Tests -119 to -142 With 1500F Kill



High Pressure Earth Storable Rocket Technology Program

.Correlated Trip Heat Flux
Tests -119 to -142 With 1500F Kill



High Pressure Earth Storable Rocket Technology Program

Results of Stability Calculations for the Task 4 Testbed Thruster

Chamber	-3	-2	-2
Pc, psi	100	250	500
D_c	1.71	1.71	1.71
D_{th}	0.819	0.52	0.37
CR	4	11	21
Isp (300:1), sec	330	335	338
% FFC	35%	35%	35%
$\Delta P/Pc$	0.6	0.24	0.12
ΔP (psid)	60	60	60
$V_{f\ inj}$ (fps)	94	94	94
$V_{o\ inj}$ (fps)	79	79	79
d_{comb} (in.)	.08	.08	.08
τ_f (sec)	.000070	.000070	.000070
$freq_f$	7142	7142	7142
$freq_{ox}$	7050	7050	7050
Freq 1T (Hz)	15,600	15,600	15,600
Freq 1L	5428	5428	5428
L' (in.)	4.2	4.2	4.2
N_e	92	92	92
Propellants	NTO/Hydrazine All Cases		

17.130-4/24

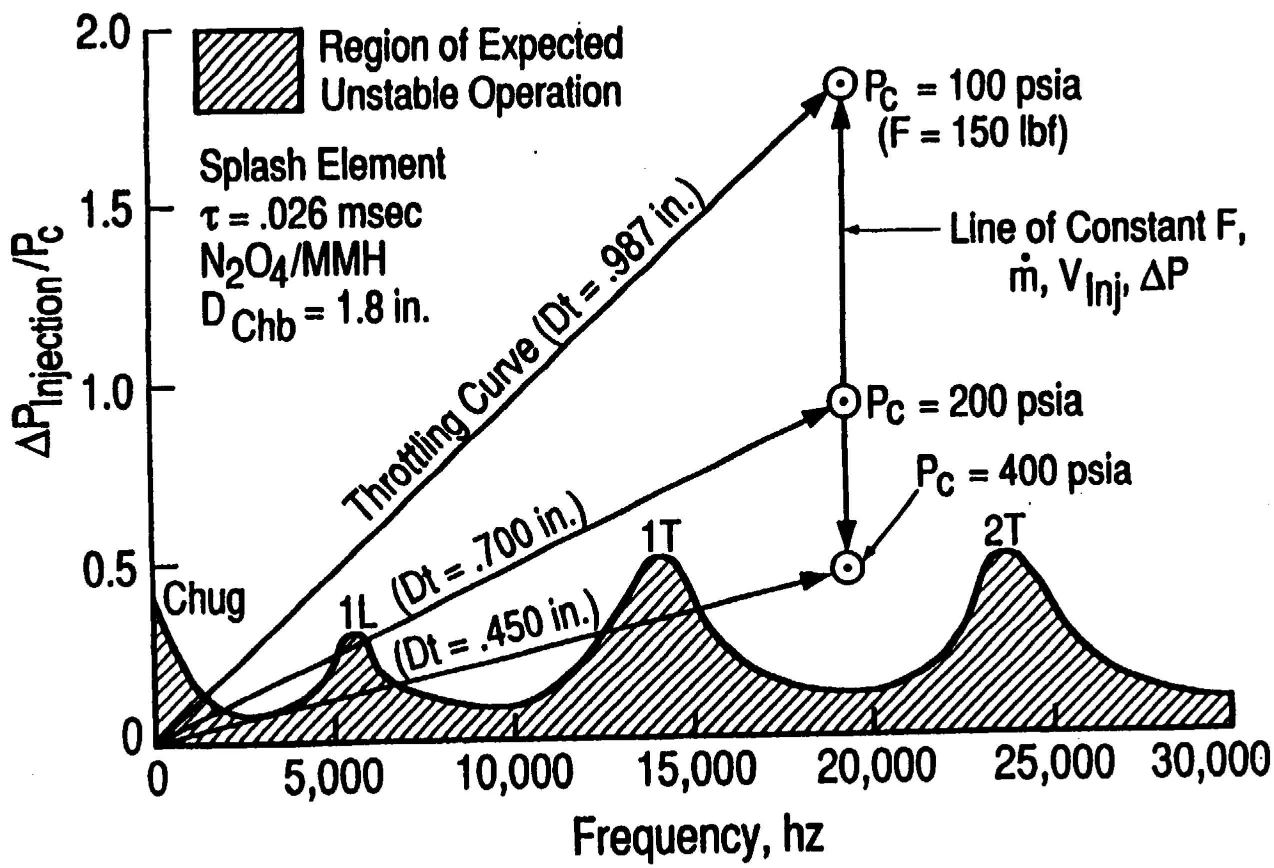
The Task 4 testbed injector incorporates splashplate elements that have been well characterized with regard to both high frequency and chug instability. Determination of its stability requires examination of chamber pressure measurements obtained with high frequency quartz pressure transducers, monitored up to their frequency limit.

The splashplate element used in the injector is well-characterized from a combustion stability standpoint. It exhibits an "injection coupling" mode of instability and, therefore, its stability characteristics are a function of its injection time lag, injection stiffness (delta P/Pc), and the acoustic resonance frequencies of the thrust chamber.

The shaded zone in the figure shows the chamber resonant frequencies and response for chug, 1L, 1T, and 2T acoustic modes (approximately 5400, 14000, and 23000 Hz, respectively). An engine operating curve which intersects the shaded zone could operate unstably at the indicated resonance, with a magnitude which depends on system damping.

High Pressure Earth Storable Rocket Technology Program

Injection Response Can Be Achieved by Throttling the Test Bed Engine



log 922.544

High Pressure Earth Storable Rocket Technology Program

TASK 4 TEST PLAN

The Task 4 testing is intended to determine the effects of increased chamber pressure on performance and heat transfer. Initial tests will explore the effect of higher trip height (better mixing) on performance, a requirement indicated by the Task 2 testing. Not all possible combinations of trip height will be tested; exact configurations to be tested will depend on the performance and thermal results.

Slow erosion of the unprotected rhenium chambers to be used for the 100 and 250 Pc is expected, even with a 'compatible' injector. In initial tests the forward portion of the rhenium chamber will be protected with Re foil, until it is determined that the trip configuration is compatible.

The 500 Pc tests will use a copper chamber because of the difficult thermal management problem at the non-optimum chamber diameter. Rhenium foil liners will also be used in this chamber to determine compatibility at 500 psia Pc.

Data to be obtained in these tests are measured and predicted altitude performance, energy release efficiency, thermal and chemical compatibility of the trip and chamber, and chamber heat transfer.

High Pressure Earth Storable Rocket Technology Program

Task 4 Test Matrix NASA Hi Pc Program Rocket Testbed

TEST NUMBER	HARD-WARE CHANGE	CHAMBER	TRIP Ht in	Lt in	PERCENT F.C.	Pc psia	MR O/F	Fvac lbf	TEST TIME sec	THROAT MATERIAL	TEST DATA MEASUREMENTS Is	C*	Twall 10 ea.	Stability
1		-2	0.05	0.55	35	250	0.80	100	10	Re	X	X	X	X
2		-2	0.05	0.55	35	250	0.95	100	10	Re	X	X	X	X
3		-2	0.05	0.55	35	250	1.10	100	10	Re	X	X	X	X
4	x	-2	0.25	0.55	35	250	0.80	100	10	Re	X	X	X	X
5		-2	0.25	0.55	35	250	0.95	100	10	Re	X	X	X	X
6		-2	0.25	0.55	35	250	1.10	100	10	Re	X	X	X	X
7	x	-2	0.05	0.75	35	250	0.95	100	10	Re	X	X	X	X
8	x	-2	0.25	0.75	35	250	0.95	100	10	Re	X	X	X	X
9	x	-2	0.25	0.75	25	250	0.80	100	10	Re	X	X	X	X
10		-2	0.25	0.75	25	250	0.95	100	10	Re	X	X	X	X
11		-2	0.25	0.75	25	250	1.10	100	10	Re	X	X	X	X
12		-2	0.25	0.75	25	250	0.80	100	60	Re	X	X	X	X
13		-2	0.25	0.75	25	250	0.95	100	60	Re	X	X	X	X
14		-2	0.25	0.75	25	250	1.10	100	60	Re	X	X	X	X
15	x	-3	0.05	0.55	35	100	0.95	100	10	Re	X	X	X	X
16	x	-3	0.05	0.75	35	100	0.95	100	10	Re	X	X	X	X
17	x	-3	0.05	0.55	25	100	0.95	100	10	Re	X	X	X	X
18	x	-3	0.05	0.75	25	100	0.95	100	10	Re	X	X	X	X
19	x	-3	0.05	BEST	BEST	100	0.95	100	60	Re	X	X	X	X
20	x	-1	0.25	0.75	35	500	0.8	100	10	Cu	X	X		X
21	x	-1	0.25	0.55	35	500	0.8	100	10	Cu	X	X		X
22	x	-1	0.4	0.55	35	500	0.8	100	10	Cu	X	X		X
23	x	-1	BEST	BEST	35	500	0.95	100	10	Cu	X	X		X
24		-1	BEST	BEST	35	500	1.1	100	10	Cu	X	X		X
25		-1	BEST	BEST	35	500	0.95	100	60	Cu	X	X		X
							CUMULATIVE TIME=		500	SEC				

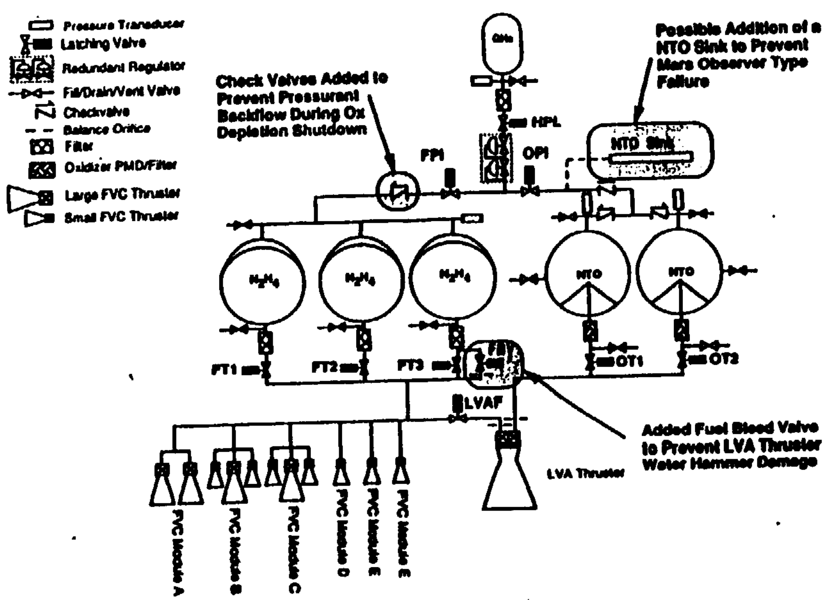
High Pressure Earth Storable Rocket Technology Program

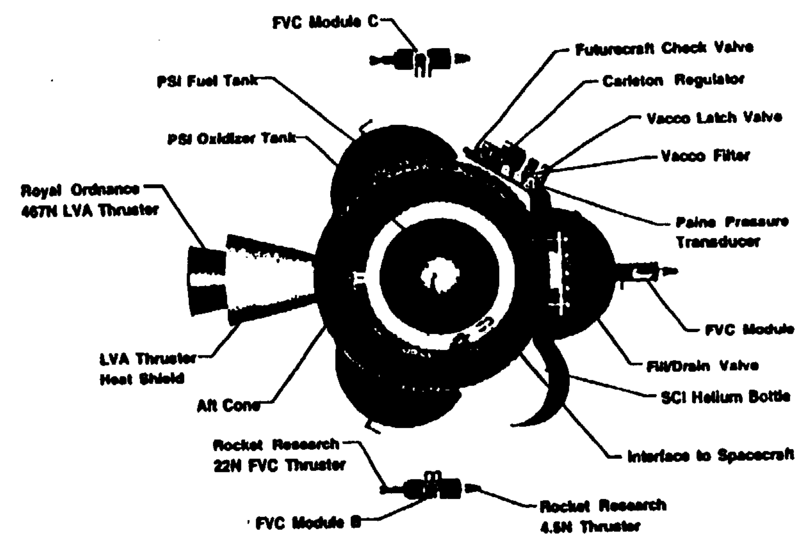
APPLICATION UPDATE

The NEAR propulsion system is representative of the Discovery class mission size; it is 1/3 to 1/5 the size of spacecraft considered prime candidates for high Pc thrusters. Even with its small amount of propellant, a significant system improvement can be realized by replacing the LEROS 1 NTO/hydrazine 100 psia thruster with a 250 Pc thruster. The 250 Pc can be obtained in this system working within the pressure limits of the existing propellant tanks. The higher pressure operation results in either 14 kg increased payload or added propellant to increase the on-orbit life by 3 months.

High Pressure Earth Storable Rocket Technology Program

APL/NEAR Space Craft Performance Would Be Improved by Hi PC 100# Thruster



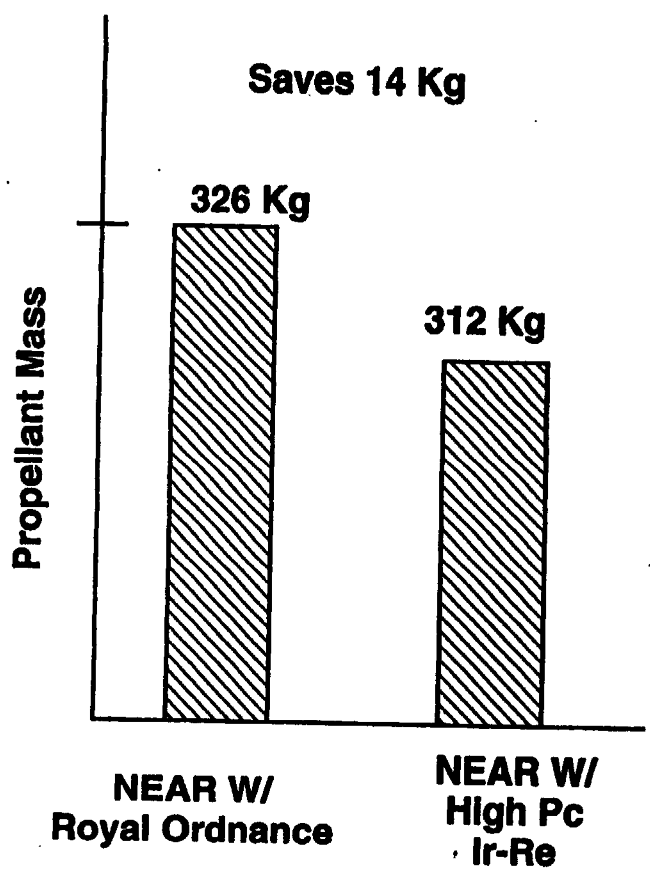


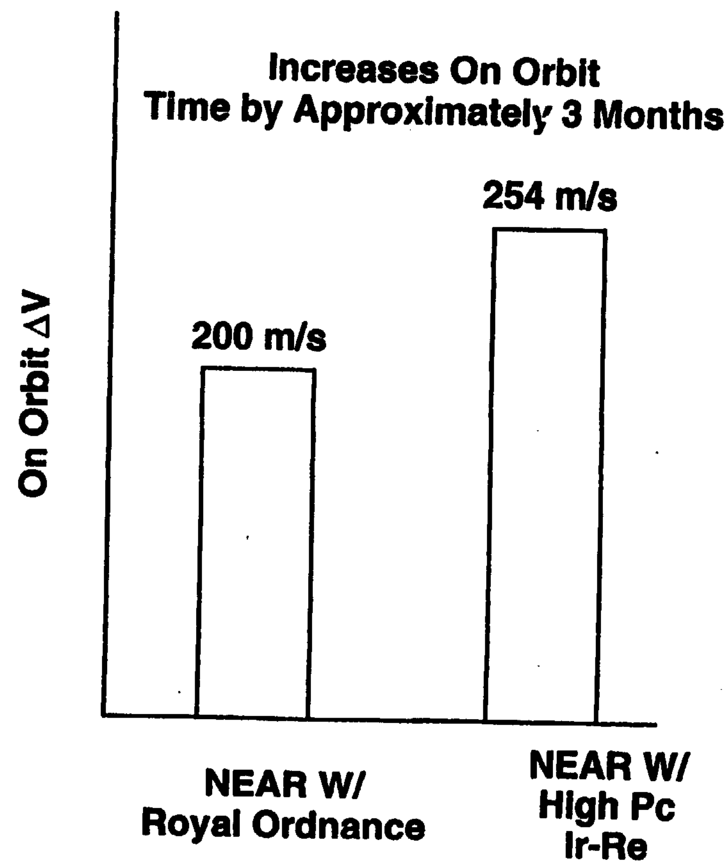
NEAR Propulsion System

Replacing LEROS 1 LVA Thruster With 250 Pc Ir-Re Thruster Provides 14 Kg Added Payload or 3 Months More Orbit Time at EROS While Working Within P/S Pressure Capability

High Pressure Earth Storable Rocket Technology Program

The High Pc Ir-Re Thruster Would Improve the NEAR Spacecraft Operation



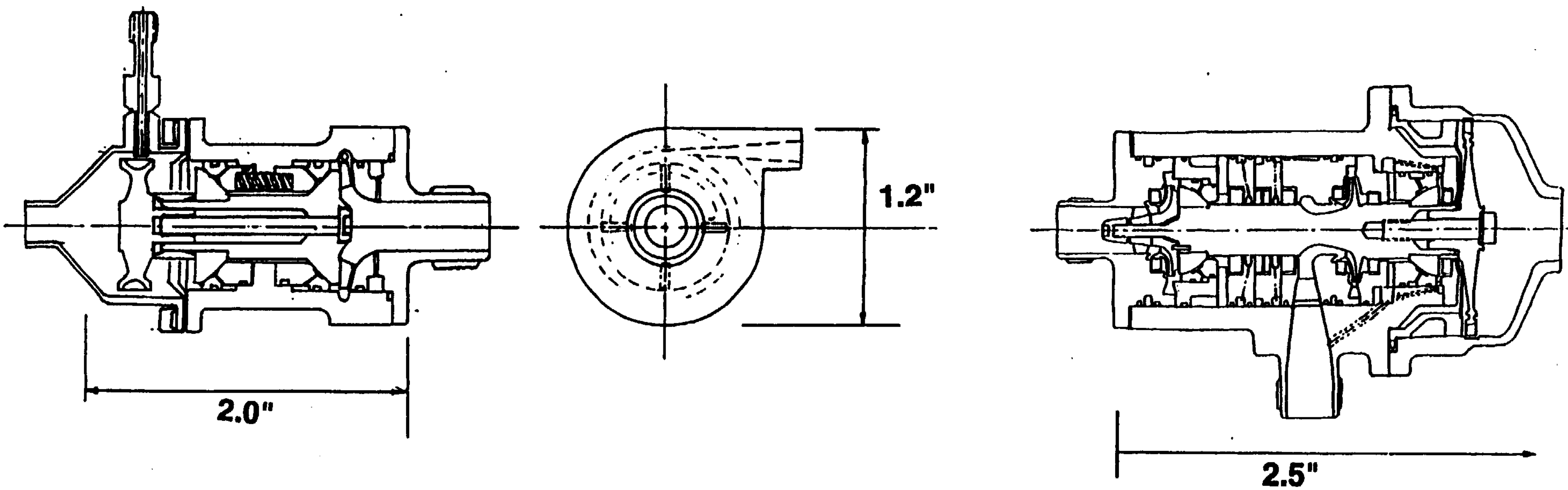


7.81.E9.2

Technology development of very small high pressure turbopumps is underway. Aerojet will be developing the pump on the left for SDI actuator applications. Two of these pumps, with the the turbine drive replaced by an electric motor operating at lower speed, would be suitable for the 500 psi thruster. The pump on the right is a concept for a bipropellant pump for a larger thruster. Replacing the turbine with a lower speed electric motor would provide a pumping system suitable for spacecraft operation. Note that the hard copy drawings are full size.

High Pressure Earth Storable Rocket Technology Program

Small Pumps for Other Systems Could Be Adapted to Hi Pc Thruster



Pump for Hydraulic Actuator

Hp	15
Δp (psi)	3,150
Q (gpm)	2,0
N (rpm)	200,000

Pump for Storable Propellant Rocket Engine

Hp	8.0
Δp (psi)	1,000
Q (gpm)	3.88/3.16 (F/O)
N (rpm)	130,000

Replacing Hot Gas Turbine Drive With Electric Motor Will Give a Pump Suitable for Large Spacecraft Application

7.81.ES.2

High Pressure Earth Storable Rocket Technology Program

TASK 3--FABRICATION

High Pressure Earth Storable Rocket Technology Program

Task 3 Fabrication

- Hardware Quantities/Suppliers

Component	Quantity	Supplier	Comments
• Valve	1	Moog	Existing
• Injector	1	Aerojet	–
• Chamber	2 (1 ea at Pc = 100 and Pc = 250 psia	Ultramet	10 Weeks Lead Time
• Trip	3 (1 ea at .05, 0.25 and 0.4 in.)	Johnson-Mathey or Englehart	–
• Trip Housing	2 (1 ea at 0.55 and 0.75 in. Length)	TECMA	–
• Ring	4	–	Existing
• Adapter	1	–	Existing
• Tungsten-Rhenium Thermocouples	30	Omega	
• Film Cooling Inserts	3 Sets	TECMA	
• Rhenium Foil	AR	Rhenium Metals	
• Platinum Foil	A/R	J-M	

17.130-4b/rt/7

High Pressure Earth Storable Rocket Technology Program

Task 3 Fabrication (cont.)

<u>Chambers</u>

- Sent RFQ to 6 Potential Suppliers:
  - Ultramet
  - Sandvik
  - Northwest Ind, Inc.
  - General Plasma
  - Applied Coatings
  - Electroformed Nickel
- Response Provided By: Ultramet, Sandvik, and Northwest Ind.
  - Northwest Ind Quoted Machining Operation Only (Aerojet to Supply Bar Stock)
- RFQ Work Statement:
  - Pricing for:
    1. 1 each and 2 each of P/N 1208173-1, -2, and -3
    2. Cost Impact of Reducing Wall Thickness From 0.190 to 0.100 in.
    3. Other Tasks: Dye Penetrant Inspection, Proof and Leak Test
  - Define Lead Time
  - Define Fabrication Process

17.130-4b/rt/1

High Pressure Earth Storable Rocket Technology Program

Task 3 Fabrication (cont.)

Chambers (cont.)

- **Ultramet Quote**
- **Unit Price, $**

Unit Price, $	Wall Thickness = 0.190		Wall Thickness = 0.100	
	1 Unit	2 Units	1 Unit	2 Units
P/N 1208173-1	24,401	23,180/each	16,307	15,490/each
-2	25,376	24,110/each	16,877	16,030/each
-3	26,367	25,040/each	17,465	16,590/each
-1, -3, and -3	70,000	66,500/set	46,600	43,500/set

- **Other Costs**:
  - Fab Tooling: $7,500
  - Test Tooling: $7,500 Maximum
- **Lead Time**: 1st Unit at 10 Weeks
  Subsequent Units at 1 Unit/Week Thereafter
- **Fabrication Process**: CVD Over Molybdenum Mandrel; Machine to Print
- **Alternate Process**: A New Proprietary Process (Not Defined) That Would Reduce Chamber Unit Cost to ~ $10,000

17.130-4b/rt/2

High Pressure Earth Storable Rocket Technology Program

Task 3 Fabrication (cont.)

Chambers (cont.)

- **Sandvik Quote**
  - **Unit Price, $**

Unit Price, $	Wall Thickness = 0.190	Wall Thickness = 0.100
P/N 1208173-1	26,930/each	Add 3-5%
-2	30,110/each	
-3	36,725/each	

- **Other Costs:** Inspection and Testing: 900 for Initial Unit and 600/each Thereafter
- **Lead Time:** 1st Unit at 12-14 Weeks; Subsequent Units at 2-3 Weeks/Unit
- **Fabrication Process:** Powder Metallurgy Plus HIP Procedure; EDM and Grind to Print

17.130-4b/rt/3

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY (*Leave blank*)

2. REPORT DATE
October 1997

3. REPORT TYPE AND DATES COVERED
Final Contractor Report

4. TITLE AND SUBTITLE
High Pressure, Earth-Storable Rocket Technology
Volume 2—Appendices A and B

5. FUNDING NUMBERS
WU–242–70–01
NAS3–27003

6. AUTHOR(S)
D.M. Jassowski

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
Aerojet
P.O. Box 13222
Sacramento, CA 95813–6000

8. PERFORMING ORGANIZATION REPORT NUMBER
E–9400

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135–3191

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
NASA CR–195427/VOL2

11. SUPPLEMENTARY NOTES
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12a. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified - Unlimited
Subject Category: 20 Distribution: Nonstandard

This publication is available from the NASA Center for AeroSpace Information, (301) 621–0390.

12b. DISTRIBUTION CODE

13. ABSTRACT ***(Maximum 200 words)***

The effect of elevated chamber pressure on combustion efficiency and heat transfer has been determined at the 100 lbf (445 N) thrust level for nitrogen tetroxide propellants. Measurements were made up to 500 psia (3.45 MPa) with testbed hardware; tests at 100 psia (0.690 MPa) and 250 psia (1.72 MPa) were made with radiation-cooled rhenium chambers. The first task of the program served to determine desirable thruster applications and operating conditions: high total impulse, i.e. communication satellite or spacecraft bus axial engines, at chamber pressures up to 250 psia (1.72 MPa) pressure-fed, or up to 500 psia (3.45 MPa) pump-fed. The hardware modifications and testing required to obtain the data were determined in Task 2, which included design-support hot fire tests; supplemental hardware, including a 250 psia (1.72 MPa) Pc rhenium chamber and a 20% fuel-film cooled platelet injector was fabricated in Task 3. Testing showed that satisfactory operation of Ir-Re radiation chambers is assured at pressures up to 250 psia and may be possible up to 500. The heat transfer data obtained show good correlation with throat Reynolds number and are generally under values given by the simplified Bartz equation; chambers equilibrium temperatures match predicted values. Preliminary optimization of trip configuration and mixture ratio were made; Isp performance from thrust measurements was within 1% of predicted values. Stability, compatibility, and front-end thermal management were determined to be satisfactory.

14. SUBJECT TERMS
Rockets; Satellite propulsion; High pressure; Heat transfer; Combustion efficiency; High performance rhenium thrusters; Iridium coatings; Radiation cooling

15. NUMBER OF PAGES
204

16. PRICE CODE
A10

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	

NSN 7540-01-280-5500